NASA CR-151972

CONCEPTUAL DESIGN OF SINGLE TURBOFAN ENGINE POWERED LIGHT AIRCRAFT

By F. Samuel Snyder, C. Gene Voorhees,
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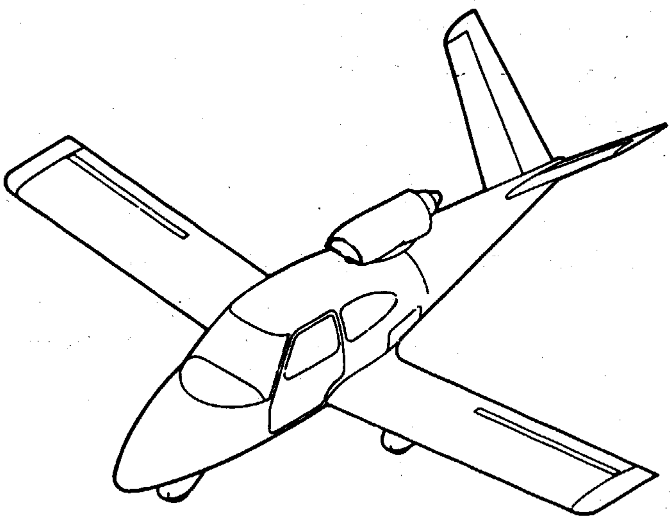
March 1977



Prepared under Contract No. NAS2-9242 by

Gates Learjet Corporation

Wichita, Kansas

for

AMES RESEARCH CENTER

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

1.0 SUMMARY

During this study, the conceptual design of a four place single turbofan engine powered light aircraft was accomplished utilizing contemporary light aircraft conventional design techniques as a means of evaluating the NASA-Ames General Aviation Synthesis Program (GASP) as a preliminary design tool. In certain areas, disagreement or exclusion were found to exist between the results of the conventional design and GASP processes. Detail discussion of these points along with the associated contemporary design methodology are presented in their respective sections of the text.

The GASP program as it was structured at the time of this study gave significantly different results in some areas when applied to the design of the present study class of aircraft. This is primarily the result of utilizing the input default values of the previous Garrett study and these differences could be mitigated somewhat by adjusting the program's input on a trial and error basis until reasonable results were achieved. Synthesis of a Learjet class aircraft using standard GASP default values yielded excellent data as might be expected since the GASP methodology and input default values were formulated from statistical cata for aircraft of this general size and performance.

Recommendations of specific areas in need of further study and reconciliation with the results of the contemporary design are outlined as well as suggestions for additional computational capabilities which would increase the usefulness of the GASP program.

The only significant new technology requirements identified with this class of aircraft were those associated with reducing the cost of the turbofan powerplant to a position more competitive with reciprocating engines.

2.0 INTRODUCTION

In a previous study (Reference 1) sponsored by NASA, the Garrett Corporation investigated the applications of small turbofan engines to single engine light aircraft. That study, in addition to engine cycle and design analysis, evaluated the total aircraft design and mission by using the NASA Ames general aviation computer-aided design program (GASP). During that study, several questions were raised concerning the configuration arrangement, aircraft performance, and modeling characteristics of the GASP program.

The objectives of the present study were to thoroughly analyze this class of aircraft by performing a conventional design process utilizing contemporary light aircraft design techniques; to evaluate the applicability of using the GASP aircraft synthesis program as a preliminary design tool; and to identify unique design and technology requirements involved. To achieve these objectives a four place utility configuration was chosen for study by conventional methods, utilizing essentially the same performance requirements outlined in the Garrett study (Reference 1). Parametric performance analyses were carried out using contemporary techniques along with the GASP program, for the purpose of validating the GASP results.

In the course of comparing the results of the two methods, emphasis has been placed on identifying exclusions and discrepancies in the GASP results to aid in possible future modification by the program authors.

3.0 LIST OF SYMBOLS

A.C.	Aerodynamic Center
AMPR	Aircraft Manufacturer's Production Responsibility
AR	Aspect Ratio
a_t	Tail Lift Curve Slope
a_w	Wing Lift Curve Slope
c.g.	Center of Gravity
c_{h_α}	Hinge Moment Variation with Angle of Attack
c_{h_δ}	Hinge Moment Variation with Elevator Angle
C_L	Lift Coefficient
$C_{L_{MAX}}$	Maximum Value of the Airplane Lift Coefficient
$c_{l_{max}}$	Maximum Value of the Airfoil Two Dimensional Lift Coefficient
C_M	Three Dimensional Moment Coefficient
c_{mac}	Airfoil Two Dimensional Moment Coefficient
$\frac{d\beta}{d\alpha}$	Variation of Upwash with Angle of Attack
$\frac{d\epsilon}{d\alpha}$	Variation of Downwash at the Tail with Angle of Attack
FAR	Federal Aviation Regulations
FOD	Foreign Object Damage
GLC	Gates Learjet Corporation
i_t	Tail Incidence Angle
i_w	Wing Incidence Angle
l_t	Distance between Wing and Tail Aerodynamic Centers

M	Pitching Moment
MAC	Mean Aerodynamic Chord
q	Dynamic Pressure
R_C	Rate of Climb
S	Wing Area
S_H	Equivalent Horizontal Tail Area
SLST	Sea Lvel Static Thrust
S_{VEE}	Total Vee Tail Area
t/c	Airfoil Thickness Ratio
$\overline{V}$	Tail Volume Coefficient
V_C	Cruise Speed
V_M	Top Speed
V_S	Stall Speed
w_f	Local Fuselage Width
W_G	Gross Weight
x	Distance Along Fuselage
α	Angle of Attack
α_{OL}	Zero Lift Angle of Attack
Γ	Dihedral Angle
δ	Control Surface Deflection
η_t	Ratio of Dynamic Pressure at the Tail to the Free Stream Pressure
τ	Elevator Effectiveness

4.0 THE CONVENTIONAL PRELIMINARY DESIGN PROCESS

The procedures followed by a designer in the conventional preliminary design process tend to be individualized and intuitive; further, they depend on the scope and magnitude of the particular project. An idealized procedure for a completely new design is given here, broken down into steps for clarity and discussion. They are as follows:

1. Establishment of design requirements and constraints. - For an airplane specification to be complete, it must include the design payload, cruising speed, altitude and range, and the takeoff and landing distances. Alternately, a stall speed may be specified which serves as an indirect specification of the takeoff and landing distances. If any of these items are omitted the designer must provide them based on his own experience and judgement to assure the viability of the final design. Of course many additional requirements and constraints are applied, ranging from FAA regulations through company policies and practices (stated and unstated), to the prejudices of the particular designer.

2. Layout of passenger and payload space requirements. - The external envelope of the aircraft fuselage must be minimized for good performance, conversely, the interior must be roomy enough for comfort. The instruments and controls must be located for good visibility, easy reach and operation and in a logical arrangement. The compromises involved may be shifted a different way for each different model, even within a given company line, in an attempt to satisfy the requirements of particular market targets.

3. Layout of the initial airplane configuration. - The designer initially assumes sizes for the wing and tail surfaces based on typical wing loadings and the expected gross weight; he then builds up a configuration around the

passenger/payload space. This may require several iterations before he is satisfied.

4. Estimation of the component weights and shift of the configuration for proper balance. - A component, such as the powerplant or battery, may be shifted, or the wing, along with the tail and main gear, may be moved to bring the c.g. into the proper range on the wing.

5. Performance estimation, parametric studies and modification of the design to meet the requirements. - Usually some level of parametric study is required to size the wing and/or the powerplant; the extent of the study depends on the degree of departure from past configurations. In some cases the entire project is an outgrowth of an extensive parametric study, where a particularly promising configuration was found.

6. Stability estimation and modification of the design as required. - Normally only the static stability is calculated at this stage to size the tail surfaces and establish the allowable c.g. range. Dynamic stability analyses, if any are planned, would be run later in the program.

7. Preliminary loads calculations. - Very rough estimates of the loads are made at this time as detailed airloads are only calculated after the design is frozen.

8. Layout of the structural arrangement and modification of the design as required. - The major structural elements must have simple straightforward load paths that do not interfere with the passenger/payload accomodations. Major elements such as spars, stringers, and fittings are roughly sized at this stage.

9. Systems layout. - Controls, electrical, fuel, hydraulic, and heating and ventilation systems, as required, are laid out at this state to assure simple systems without interference.

10. Cabin mock-up construction. - A mock-up is used as a three dimensional engineering design tool; in addition it also serves as a sales tool for management. This is usually the first tangible item presented to management and the occasion for the first feedback.

11. Review with manufacturing. - The purpose of a manufacturing review of the design is to identify potential fabrication problem areas and to enlist suggestions on methods of minimizing overall costs.

12. Aircraft design report draft. - The design report summarizes the preliminary design work and gives a detailed definition of the airplane for use in the detail design phase.

These twelve steps are not an orderly, linear process as might be intimated by the above listing, but rather, a more or less simultaneous continuous process. All steps are kept in mind by the designer from the beginning, and all steps, including the first one, are subject to change as the design progresses or from management input.

5.0 DESCRIPTION OF THE GASP PROCEDURE AND COMPARISON WITH THE CONVENTIONAL DESIGN PROCESS

The General Aviation Synthesis Program (GASP), a digital computer program developed by NASA-Ames Research Center, is basically a conceptual design tool for the aircraft designer who has to investigate the interaction among the various disciplines involved in the design process - namely, aircraft geometry, performance, propulsion, structures, weight and balance, economics. federal regulations, etc. - before he arrives at the end result which presumably is the best possible compromise that meets the design goals. This is a very complex iterative process that normally takes several man-months when done manually, even with the help of all the usual design charts. The goal of the GASP is to allow the designer to carry out this task in a fraction of that time.

The program has several subroutines to carry out the analysis within each discipline and a control routine which provides the user the flexibility to call any subroutine at any stage except at the very beginning when the basic geometry and powerplant size are determined. In general, these subroutines were originally developed for purposes other than GASP. The combination of these subroutines with the control program yields a very complex computer program with over 200 input parameters, several hundred assumptions inherent in the program and at the time of this study almost negligible documentation on program usage, program logic or input definition. Thus, while it is a powerful time saving tool for the conceptual design engineer, it is difficult to use without some minimum knowledge of the internal structure of the program and documentation of methodology.

The program allows the user to select his own sequence for arriving at the final configuration. For example, he can size the powerplant, compute

the range and operating cost and then change the geometry, increase the payload, decrease the cruising speed or specify a new powerplant and look at the effect of any or all of these changes on weight, cost or performance.

While many of the discrete operations performed by GASP are similar in nature to those utilized in a contemporary design effort, on a typical run, the operational task flow is somewhat different. Starting from basic input data consisting of gross weight, payload and performance criteria, GASP determines a baseline aircraft geometry and proceeds to compute the cruise, takeoff and landing aerodynamics. The first performance calculation and test comes in the form of a landing distance calculation. If the landing criterion is not met, the program resizes the wing and loops back to the starting point (geometry determination), otherwise it proceeds to size the engines on the basis of the takeoff requirement or a cruise speed specification. Optionally the engine thrust may be specified. With the engines sized, the program then computes structural weight, balances the aircraft and flies a mission profile to compute range. If the range requirement is met, the program finishes the case by computing the cost factors. If the range requirement is not met, the program increases or decreases the gross weight and loops back thru the starting point.

in the foregoing manner, the GASP program is able to produce a solution aircraft which has been synthesized to achieve the required performance criteria within the bounds of the design and geometric constraints placed upon it. Assuming that all of the aerodynamic, propulsion, weight and performance data predicted were valid, there is no guarantee that the solution aircraft is a currently viable product as the required size of engine or some other component or system may not be in existence if the program is allowed to resize these components in order to meet performance goals. Converse to this approach, the contemporary method of design starts with existing or projected engine, assumes geometry, analyzes this baseline and perturbates about this baseline geometry. In the final step of the contemporary design process, all of the

performance constraints are viewed simultaneously to define an acceptable envelope of geometric excursion.

The final judgement outlined in the description of the contemporary design process was based on weight which translates directly to production cost, a judicious choice if a particular performance requirement is solidified. Like everything else, performance has a price and market analysis for a given year establishes the acceptable associated cost levels. In the GASP analysis, performance excursions from the baseline can be made with relative ease to evaluate the associated cost variation as will be shown later. In this particular contemporary design final analysis, performance associated costs may be viewed indirectly by noting the change in wingspan required by the desired performance. To explain, the contemporary cost analysis methodology utilizes AMPR weight as a major element in the development and production cost buildup, therefore, any factor that increases the aircraft empty weight, increases the costs. In the present study, weight variations were primarily a function of wingspan; as the span increased, the weight increased. Obviously then, there is a direct relationship between development/production costs and wingspan which allows costs associated with a particular desired performance to be assessed on the basis of the wingspan change required to achieve it.

In view of the preceding discussion, the differences in approach between GASP and the contemporary design method may be summarized by saying that GASP synthesizes a single solution to satisfy the desired performance and constraints while the contemporary method analyzes variations from a given design point to establish geometric boundaries within which satisfaction of all performance requirements is simultaneously achievable. The GASP program may be used in the conventional parametric study manner, however it requires repeated program submittals in order to build the data matrix for trade studies.

6.0 THE CONVENTIONAL AIRPLANE DESIGN

6.1 Human Factors

The passenger capsule layout is illustrated in Figure 1, the boundaries shown represent the inner walls of the cabin. The cabin volume is sized and proportioned to provide good comfort, excellent visibility and adequate baggage area. While the cabin volume shown is comparable to that of a Cessna Cardinal, it is superior to those provided by the Cessna 172, the Piper Cherokee and the Beech Bonanza. Beyond that, it is significantly more comfortable than the baseline Garrett airplane (ref. 1) which had adequate width, but was somewhat lacking in headroom and rear seat leg room. This comparison is not a criticism of the designs mentioned but simply a recognition of the fact that the projected market cost of the present design project would not tolerate less than optimum comfort or space.

The forward visibility shown in Figure 1 is superior to most single engine light planes for the simple reason that there is no engine in the nose. The lower vision angle is approximately the maximum that will allow the nose to be within the field of view of a pilot looking straight ahead; thus it provides a longitudinal and lateral attitude reference with minimal obstruction to vision. Some difficulty will be encountered with the installation of instruments and radios in the panel, particularly the longer ones, due to the slope of the cowl deck. In Figure 1 the lower vision angle represents that for a 5 percentile (short) man and the upper vision angle for a 95 percentile (tall) man.

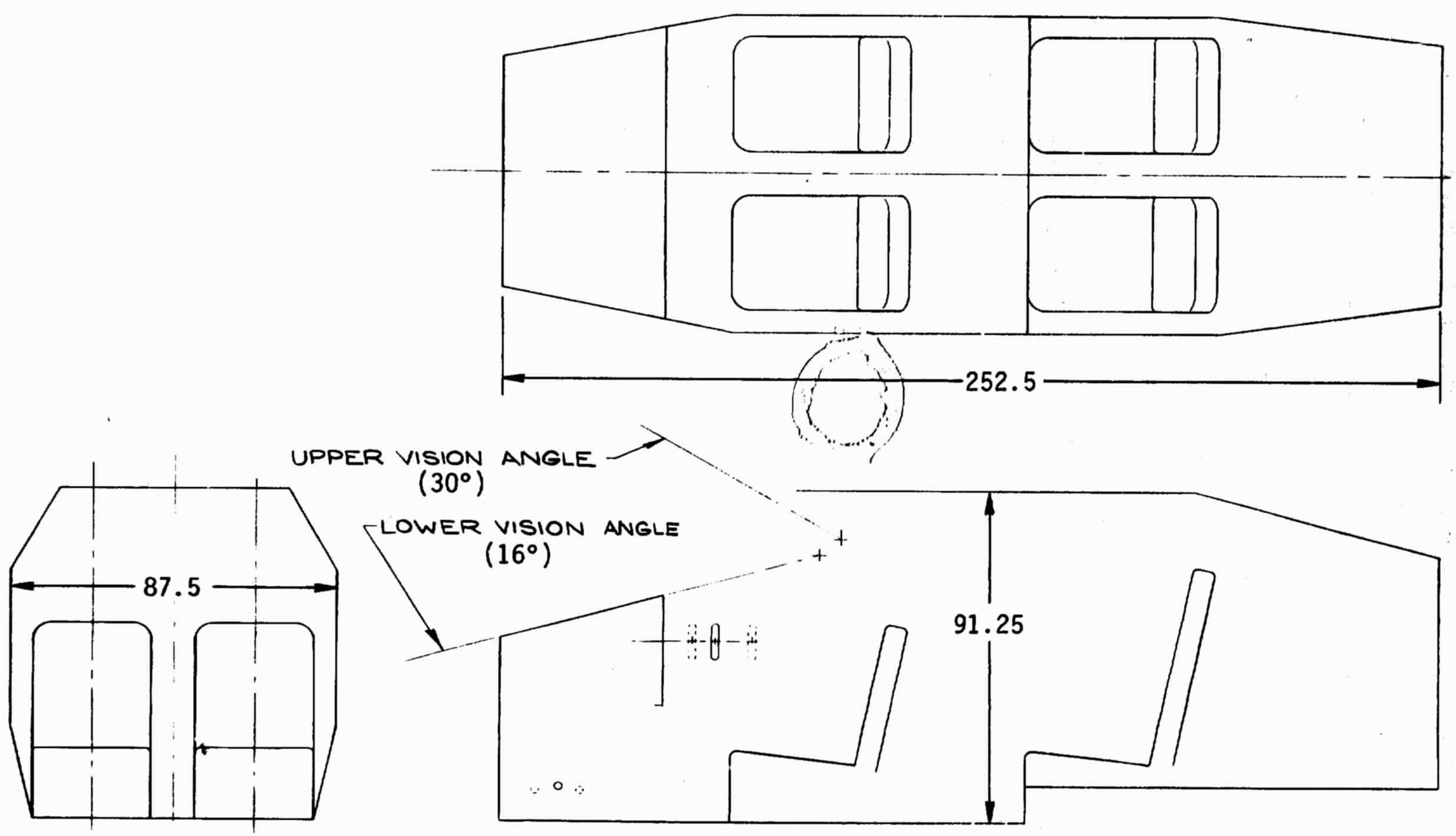

FIGURE 1 - PD1500 Cabin Layout

6.2 Airplane Configuration

The first configuration laid out, PD1501, is shown in Figure 2. This is a low wing airplane with a podded engine mounted on the aft fuselage, a vee tail, and fixed tricycle landing gear. The wing is the same size as that of the baseline Garrett airplane and incorporates the GA(W)-1 airfoil, full span Fowler flaps, and lateral control spoilers. The vee tail was selected over a twin tail for simplicity and lower parts count, and at this stage is only roughly sized.

The wing spar is located at the aft doorpost and runs under the front edge of the rear seat. The engine mount is aligned with the aft cabin bulkhead. The spring main landing gear is mounted to the aft side of the main spar carrythru. The oleo nose gear is mounted on the forward cabin bulkhead.

Two doors are provided for easy entry and exit. The step height is fixed by the clearance between the wheel and the wing necessary to allow for the landing gear stroke. With this short wing chord, there is no problem with clearance between the deflected flap and the ground.

The podded engine provides the simplest installation and affords good access for maintenance, though it also has a fairly high wetted area. The pylon length is set to the minimum that will avoid separation caused by interference between the nacelle, pylon, and fuselage.

The principal disadvantage of this configuration is its awkward appearance.

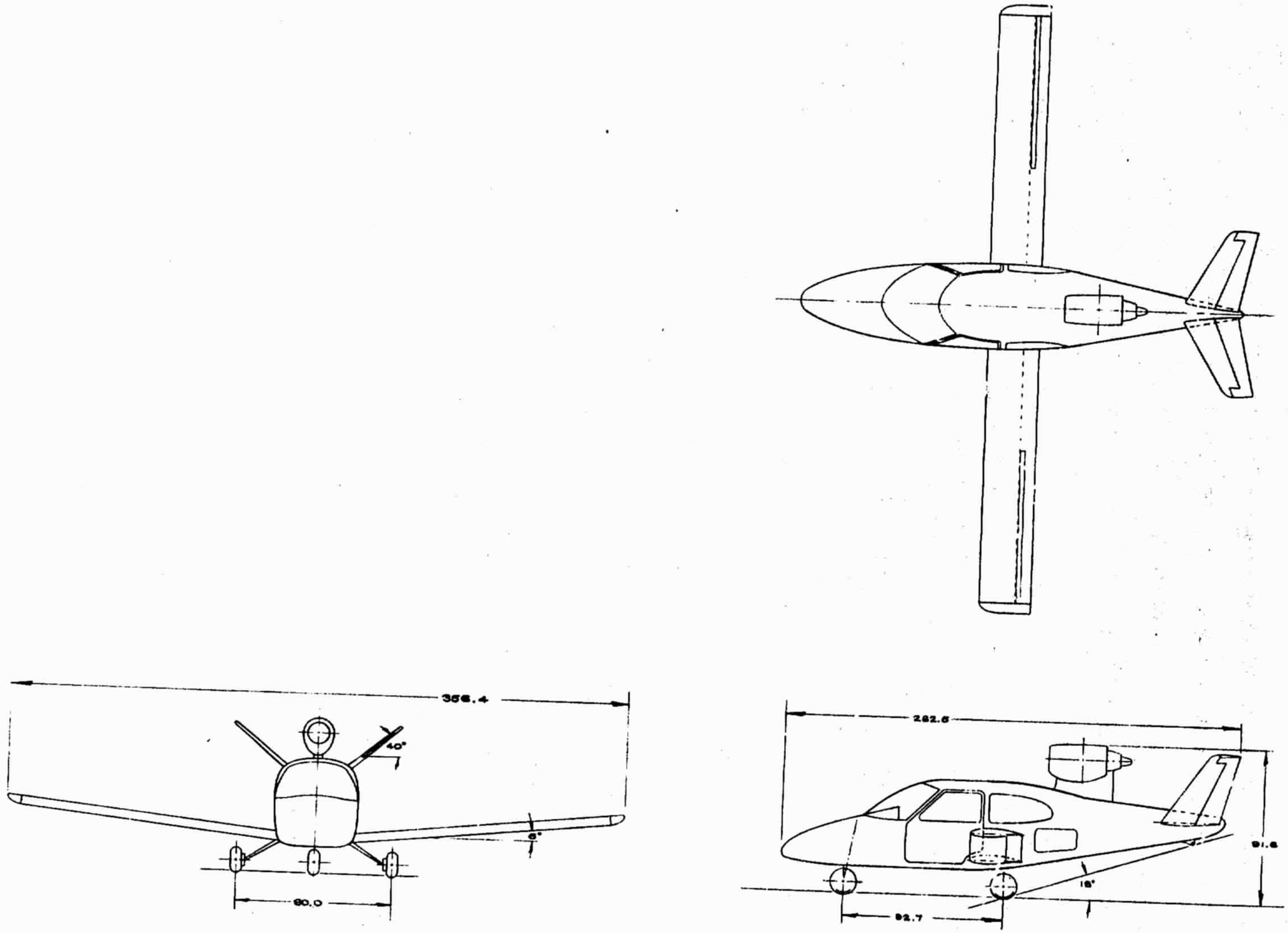

FIGURE 2 - PD1501 General Arrangement

The second configuration, PD1502, is shown on Figure 3; this features a semi-buried engine with a short S duct. This arrangement has less wetted area, but a longer intake duct. Maintenance access, particularly to the bottom mounted accessories, is more difficult.

The tail surfaces are larger than those of PD1501, reverting to the baseline aircraft's area and aspect ratio. Wheel fairings have also been added, primarily for esthetics.

Figure 4 shows the third configuration, PD1503, which has a buried engine with a bifurcated inlet duct. The high aft fuselage eliminates the need for a long tailpipe and allows the use of a conventional tail. The inlet length is limited by clearance with the door. It is not possible to put the inlets below the wing leading edge since the air would somehow have to pass through the wing structure. In addition, a lower inlet position would pick up more FOD, particularly rocks thrown up by the nose wheel from a gravel runway. As it is, the ducts of PD1503 eliminate the lower baggage compartment.

This complicated inlet ducting adds weight and is expensive to build, causes inlet distortion and increases duct losses. Access for maintenance is somewhat more difficult than a podded engine configuration.

The conventional tail on this airplane is the same size as the vee tail of the PD1502.

After study of these three configurations, the PD1502 was chosen as the most promising; the work that follows was done on that configuration.

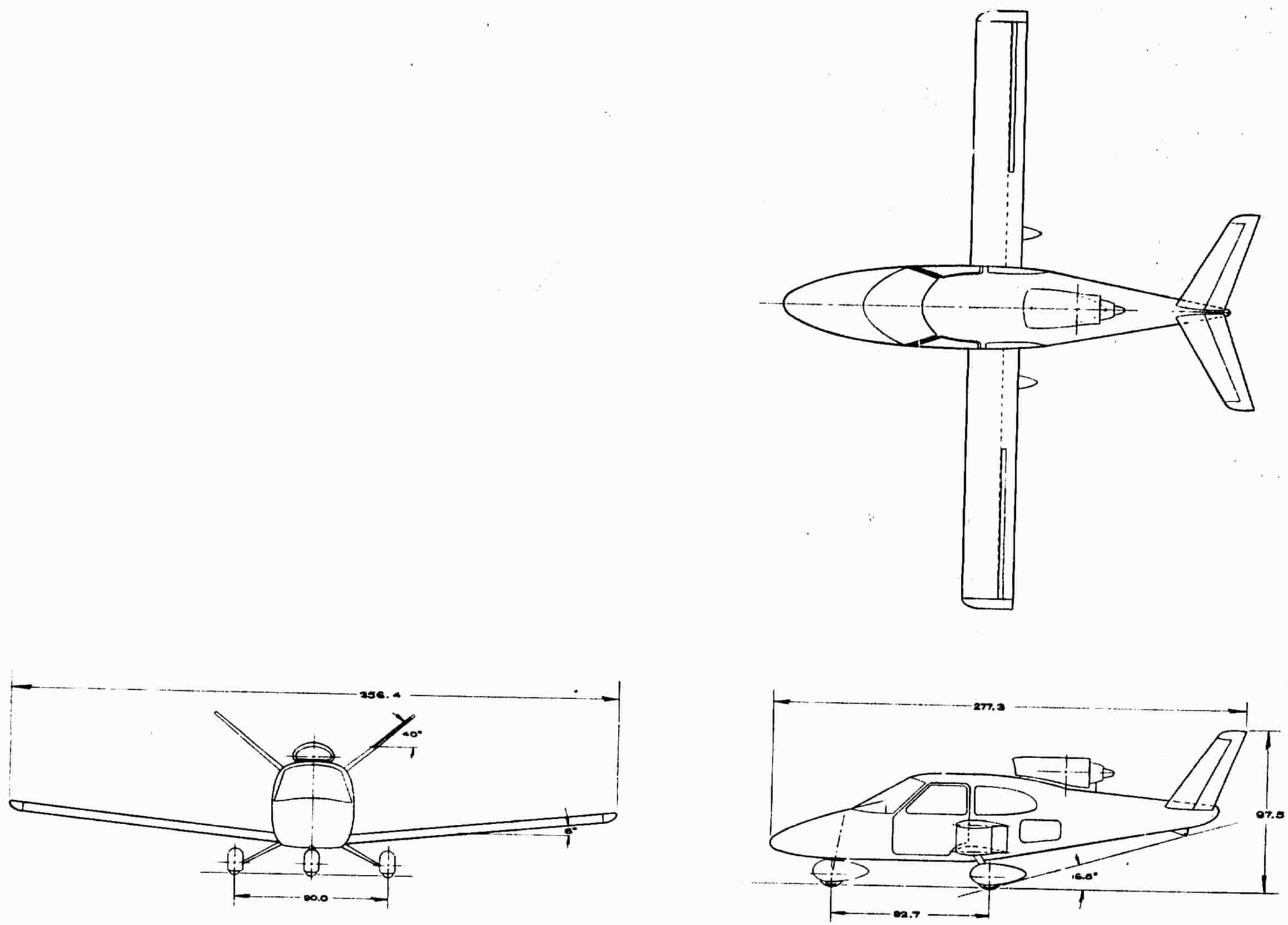

FIGURE 3 - PD1502 General Arrangement

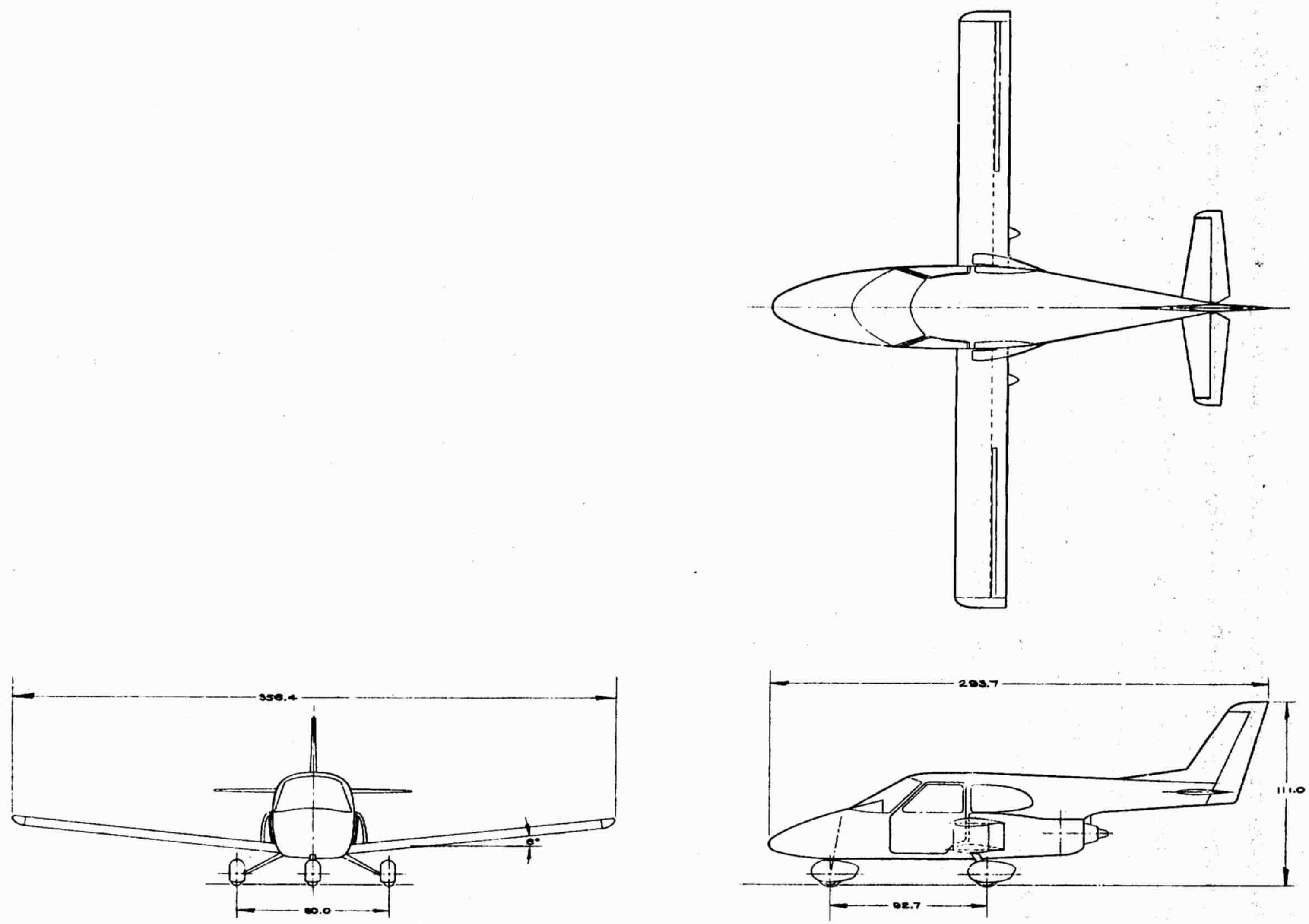

FIGURE 4 - PD1503 General Arrangement

6.3 Weight and Balance

The most accurate method of quickly estimating airframe weight is by comparison with a line of similar airplanes built by the same company. However, if the past history is sketchy or unobtainable (or not documented), or if the project at hand represents a major departure from past practice, a better method must be found. General aviation companies treat weight data as proprietary; thus, it is not widely available. Some statistical trend equations exist but must be used with care, for the reason that they may have been derived from insufficient data. Thus the weight estimation procedure becomes a combination of calculation with trend equations and comparison with past airplanes.

Table 1 gives a comparison of the PD1502 weight summary with that of the Garrett baseline airplane (Reference 1). The difference in the wing weights can be attributed to the difference in gross weight. The difference in fuselage weight is due to the larger cabin of the PD1502. Differences in landing gear and controls represent a simple disagreement. The difference in equipment weight is in the furnishings. The final difference is in the fuel quantity, which for PD1502 was increased to round off the gross weight to 907 kg (2000 lbs.).

When the balance was calculated the airplane was found to be tail heavy. In order to shift the most forward c.g. and the most aft c.g. each forward about 17% MAC, the wing, tail, and main gear were moved aft 160 mm (6.3 in.). This result is plotted on Figure 5. Three loading schedules are shown; the most forward case, the most aft case, and the maximum cabin load. While this c.g. range appears large at first glance, it must be kept in mind that with the high aspect ratio wing the chord is narrow, and the tail volume coefficient is relatively high. Whether this c.g. travel is indeed acceptable

TABLE 1

PD1502 WEIGHT SUMMARY

(Weight in kg (lb))

GROUP	PD1502	GARRETT BASELINE
Wing	98.0 (216.0)	94.8 (209)
Tail Surfaces	19.1 (42.2)	19.5 (43)
Fuselage	83.9 (185.0)	76.2 (168)
Landing Gear	49.0 (108.0)	36.3 (80)
Controls	20.4 (45.0)	15.9 (35)
Nacelle	8.0 (17.6)	8.6 (19)
Propulsion	56.2 (124.0)	52.6 (116)
Instruments	10.8 (23.7)	
Avionics	18.1 (40.0)	77.1 (170)
Electrical	22.7 (50.0)	
Furnishings	45.4 (100.0)	
Dry, Bare Empty Weight	431.6 (951.5)	
Paint	3.6 (8.0)	
Unusable Fuel	2.7 (6.0)	
Licensed Empty Weight	437.9 (965.5)	381 (840)
Payload (Design)	272.2 (600.0)	272.2 (600)
Maximum Fuel	197.1 (434.5)	181 (399)
Gross Weight	907.2 (2000.0)	834.2 (1839)

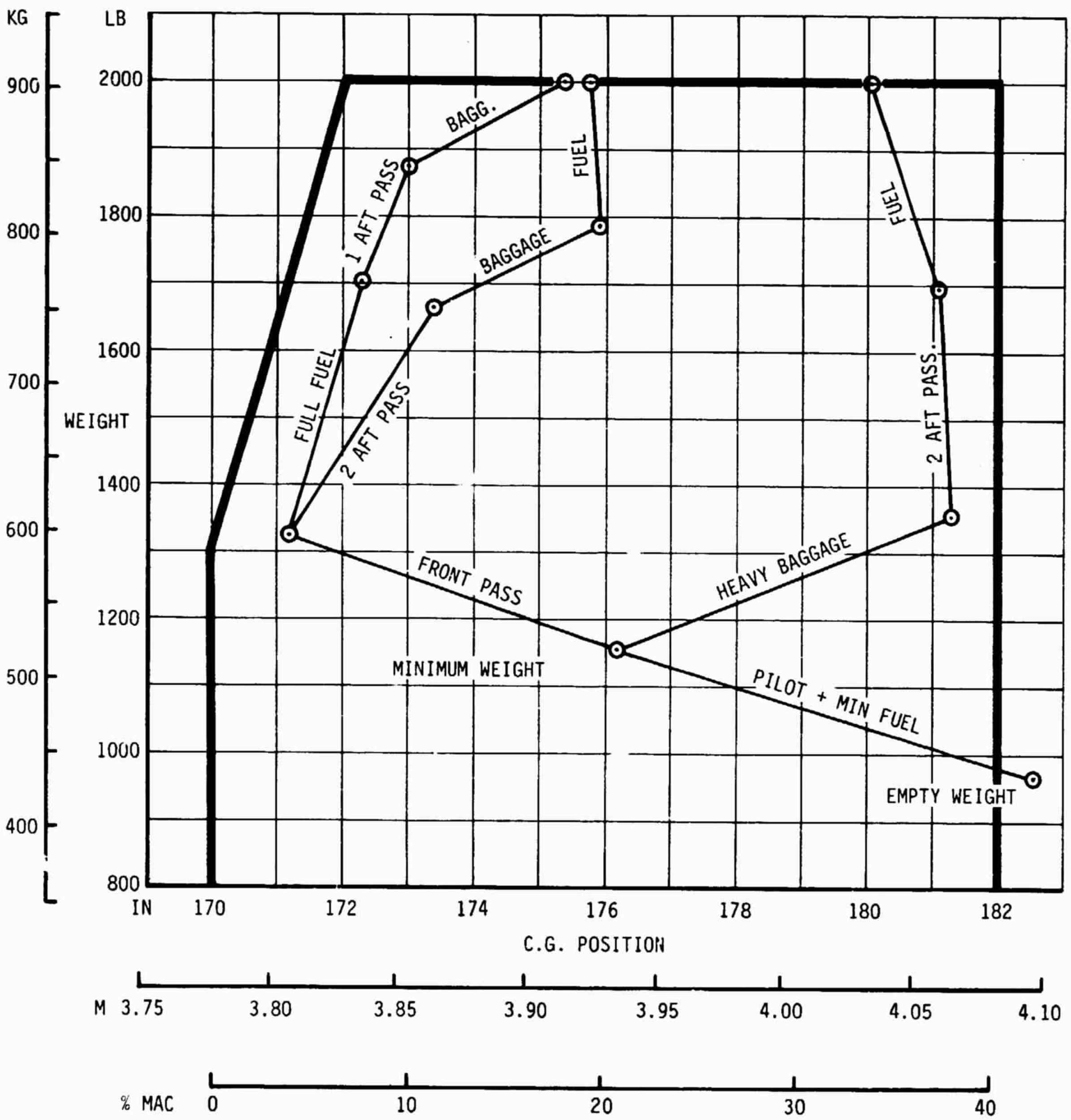

FIGURE 5 - Center of Gravity Diagram

will be determined when the stability and control calculations are made. The empty weight c.g. is outside the envelope, however this is not a flight condition and this point is ahead of the main gear, therefore, tail tipover will not be a problem.

6.4 Performance - 400 Lb. Engine

Because this is a new class of airplane, a parametric study was run over a wide range of wing spans and areas. Only constant chord planforms were considered. The NASA GA(W)-1 airfoil of 17% thickness (Reference 2) was chosen because it was felt that the final result would have a higher than usual aspect ratio and the thickness afforded by this state of the art section would aid structurally. The span was varied from 7.6 to 12.2 m (25 to 40 ft.), and the aspect ratio from 4 to 20.

A simple computer program was written to perform the required parametric looping. For each wing configuration it calculated and printed the gross weight, wing area, rate of climb, speed for best rate of climb, top speed for that thrust setting, and stall speed. For iterations at different altitudes or thrust settings, the program was rerun with appropriate inputs. Two other programs were written which combined the looping feature with takeoff and landing routines. The initial series of runs used a Garrett 1779 N (400 lbs.) sea level static thrust turbofan engine (Reference 1).

Figure 6 shows the variation of gross weight with span and area. This variation reflects the variation in wing weight alone since the rest of the airplane is held constant. This weight is used in the succeeding performance calculations, so that the effect of weight variation with configuration is accounted for. Based on a wing weight estimation procedure commonly used for this class of aircraft, it may be seen that span is the primary variable in weight, while area is a secondary variable. Note that weight decreases with increasing area at constant span. This is the opposite of what might be expected intuitively; evidently the weight of bending material decreases due to increasing thickness, faster than the skin weight increases.

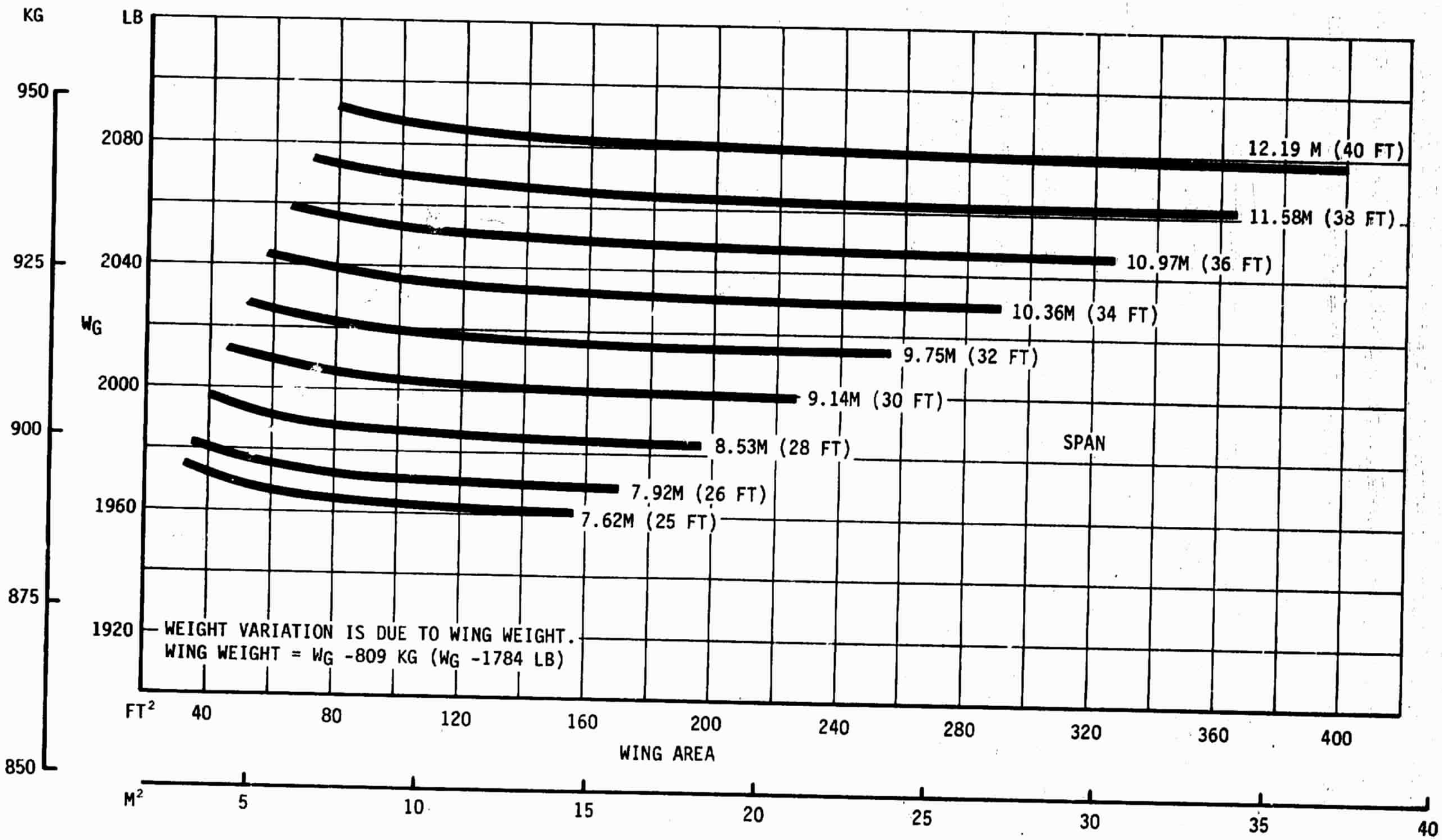

FIGURE 6 - Gross Weight

The flaps up stall speed is shown in Figure 7. While the three dimensional $C_{L_{max}}$ is calculated as a function of aspect ratio, the stall speed appears to be a function of wing area alone. Although the $C_{L_{max}}$ is higher with the greater span and aspect ratio, the associated weight increase more than compensates for it. The overall result is the stall speed of the longer spans being slightly greater than that of the shorter spans.

Figure 8 gives the sea level rate of climb. The rate of climb improves with increasing span, but the rate of improvement decreases at the higher spans. At constant span, the rate of climb improves with a decrease in area, due to the decreased wetted area and skin friction drag. The hook on the lower curves is due to the fact that the stall speed is higher than what would normally be the best rate of climb speed. The dashed line shows the FAR 23 climb requirement; the rate of climb in feet per minute must be greater than ten times the stall speed in miles per hour. Thus only those configurations to the upper right of this curve are acceptable. While this requirement is strictly applicable only for the stall speed and rate of climb in the takeoff configuration, with gear down and takeoff flaps, it is employed here as a useful guide.

The service ceiling curves of Figure 9 follow the rate of climb curves. The variation with span is greater, and the hook on the lower curves is more pronounced. Note that while the rates of climb are not exceptional, the ceilings are excellent in comparison with comparable piston powered airplanes.

Figure 10 shows the sea level top speed. This varies mainly as a function of area; span has little effect.

Figure 11 shows the sea level cruise speed; this variation is similar to that of the top speed.

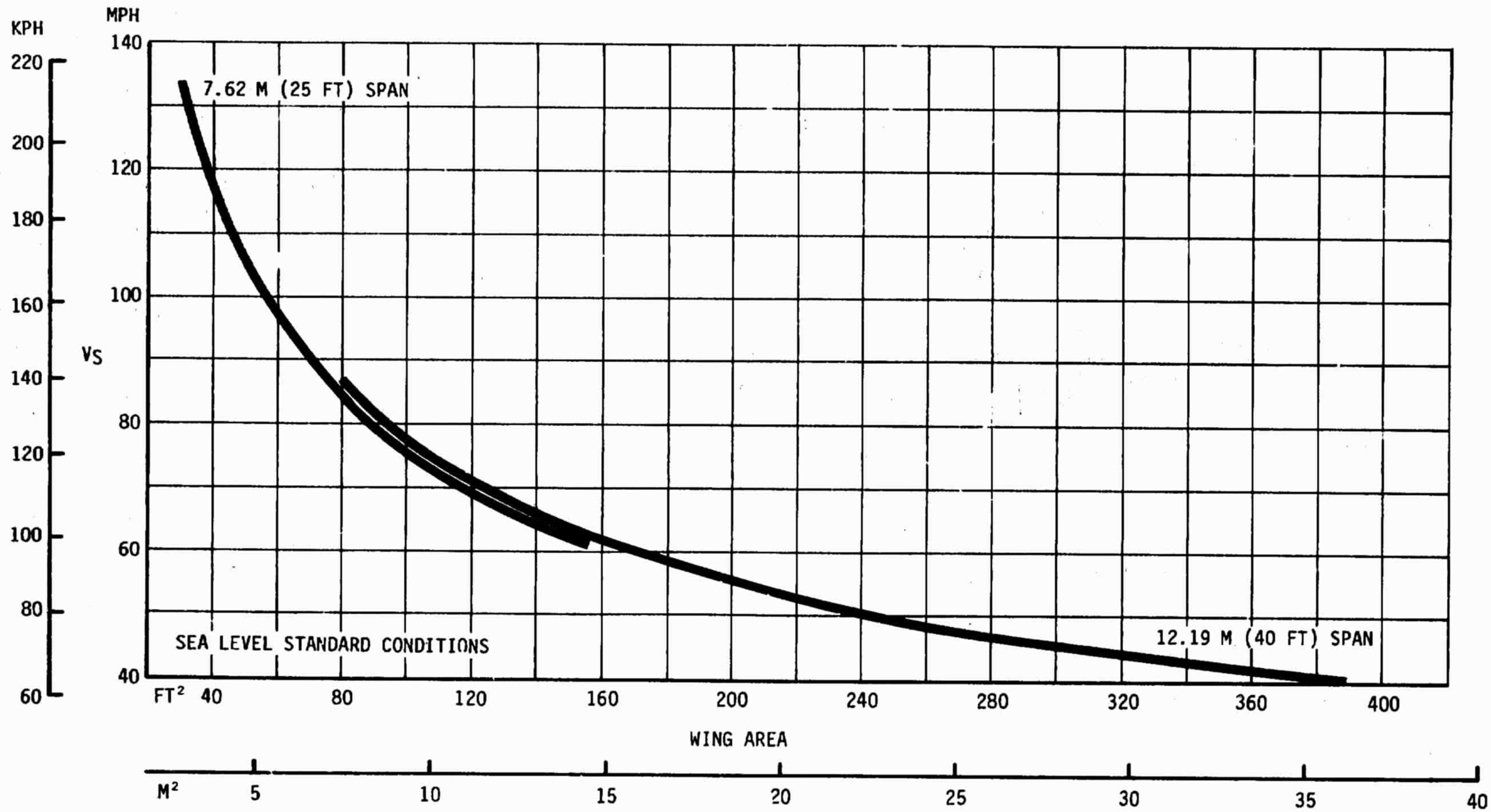

FIGURE 7 - Flaps Up Stall Speed

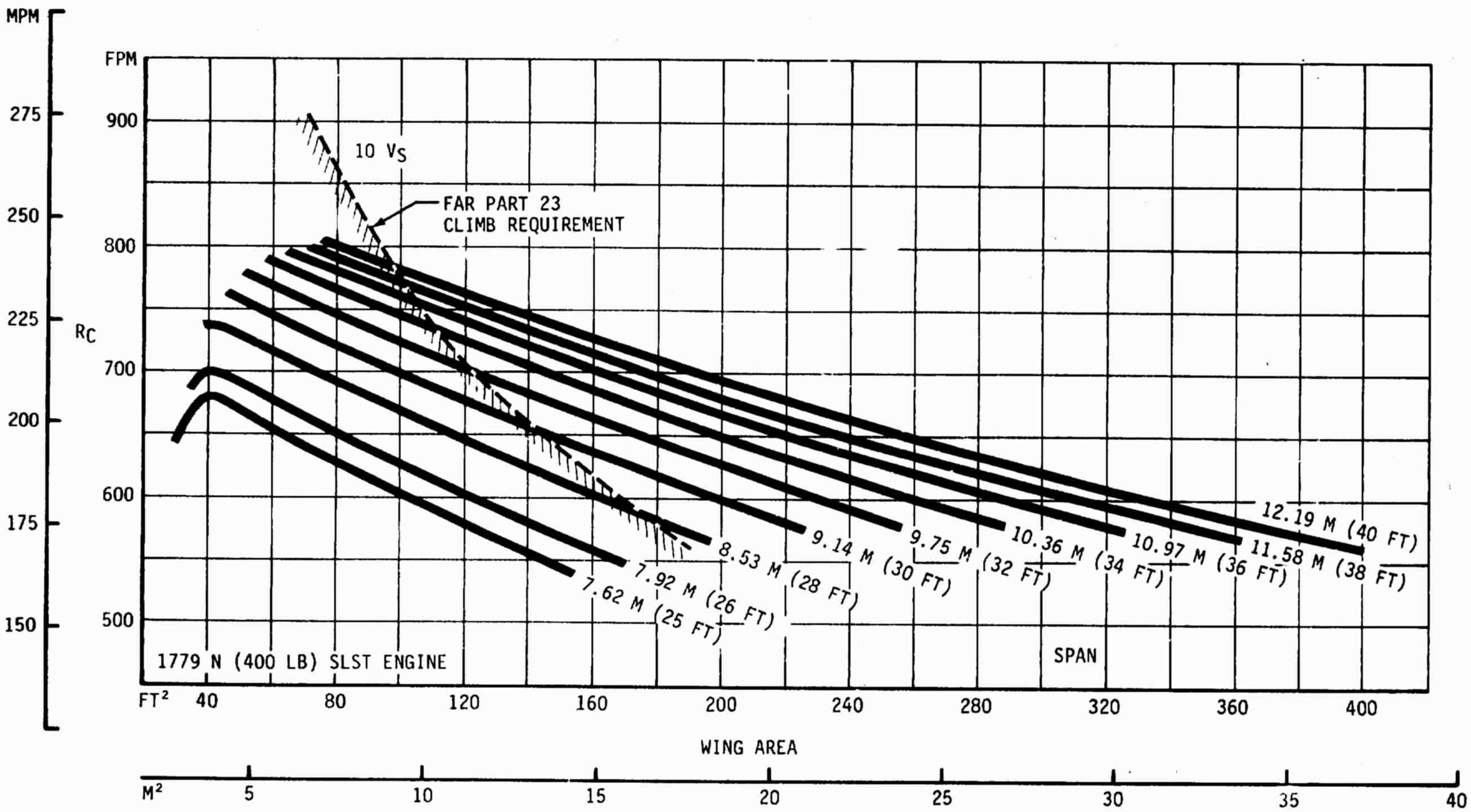

FIGURE 8 - Sea Level Rate of Climb

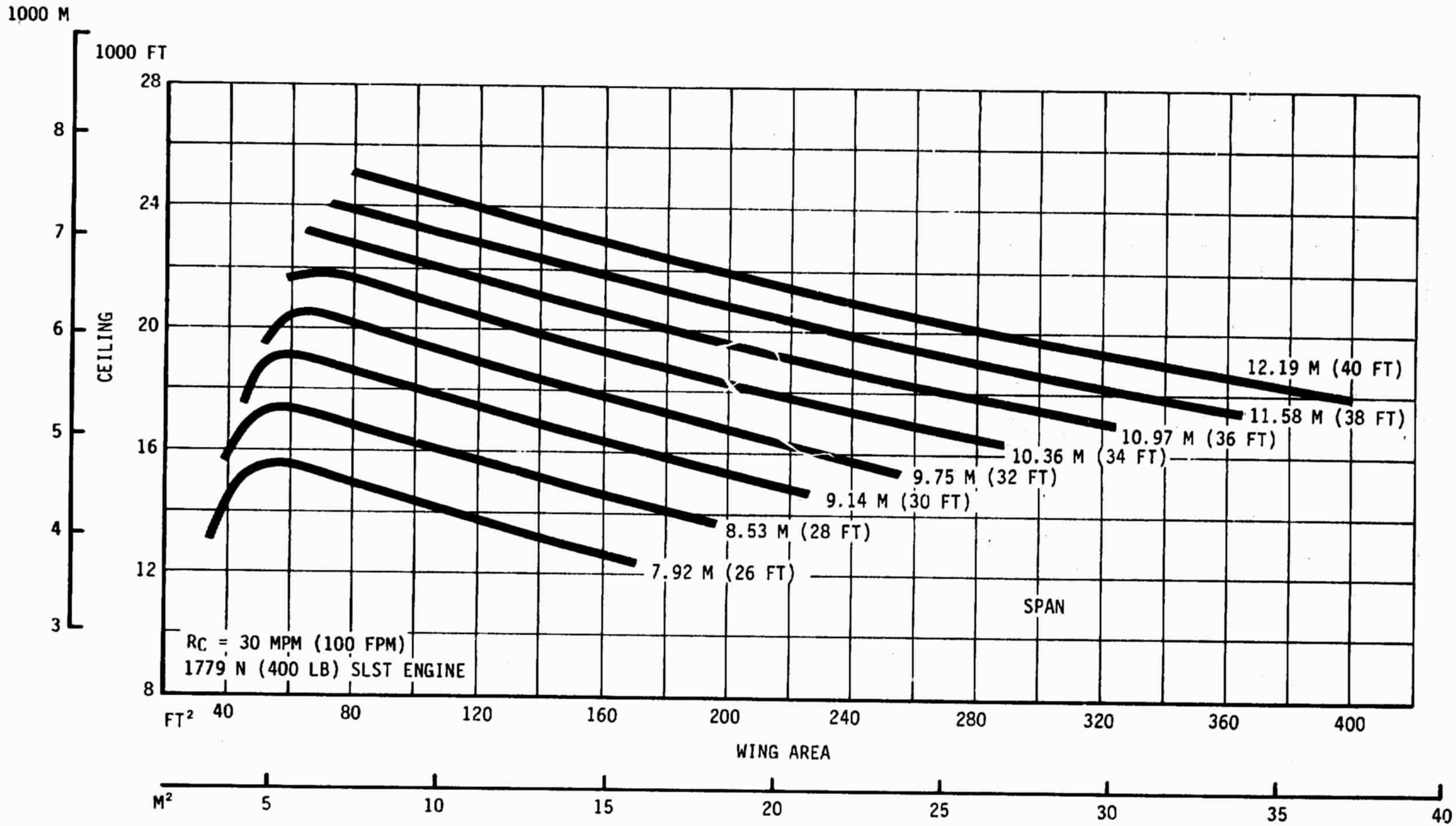

FIGURE 9 - Service Ceiling

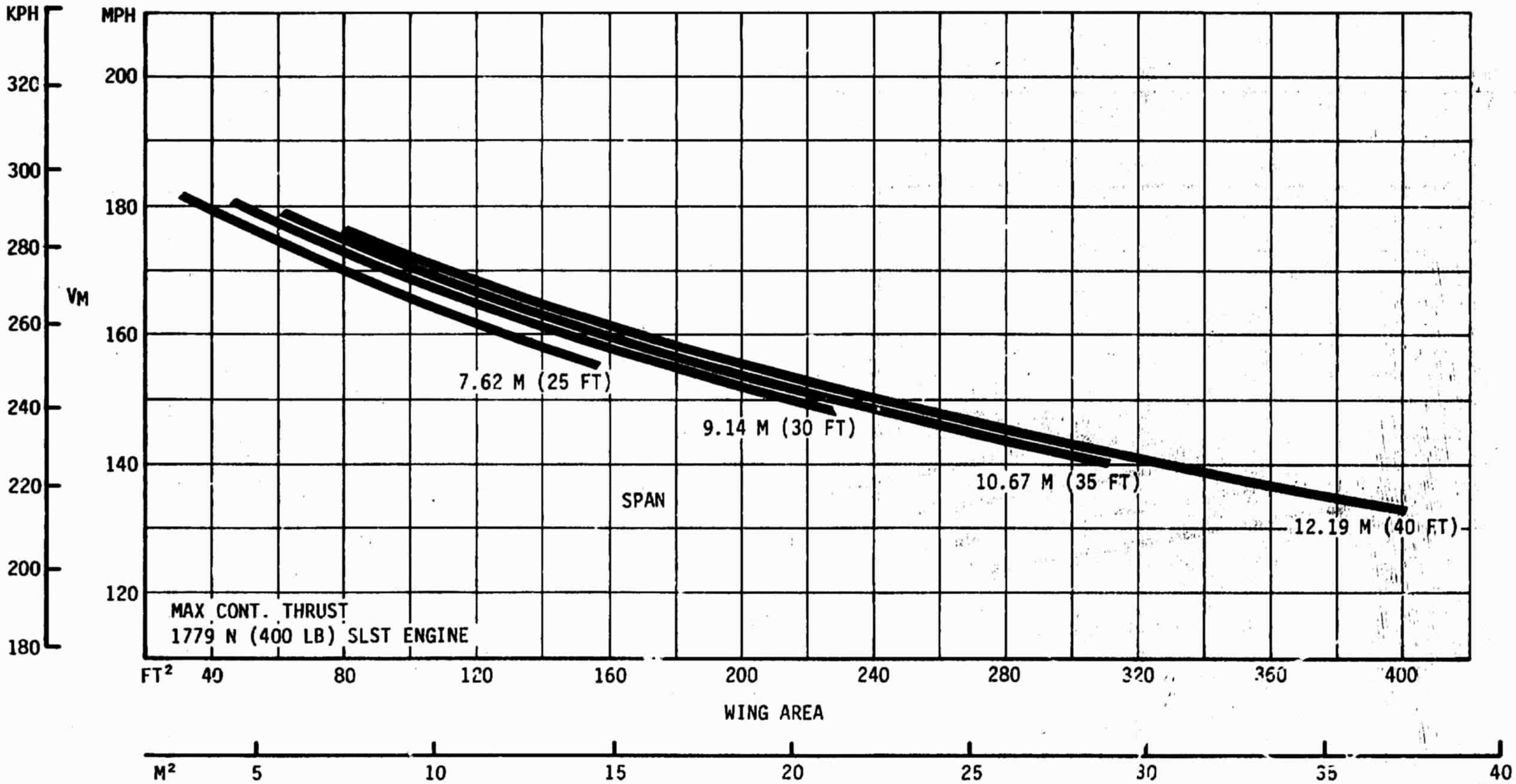

FIGURE 10 - Sea Level Top Speed

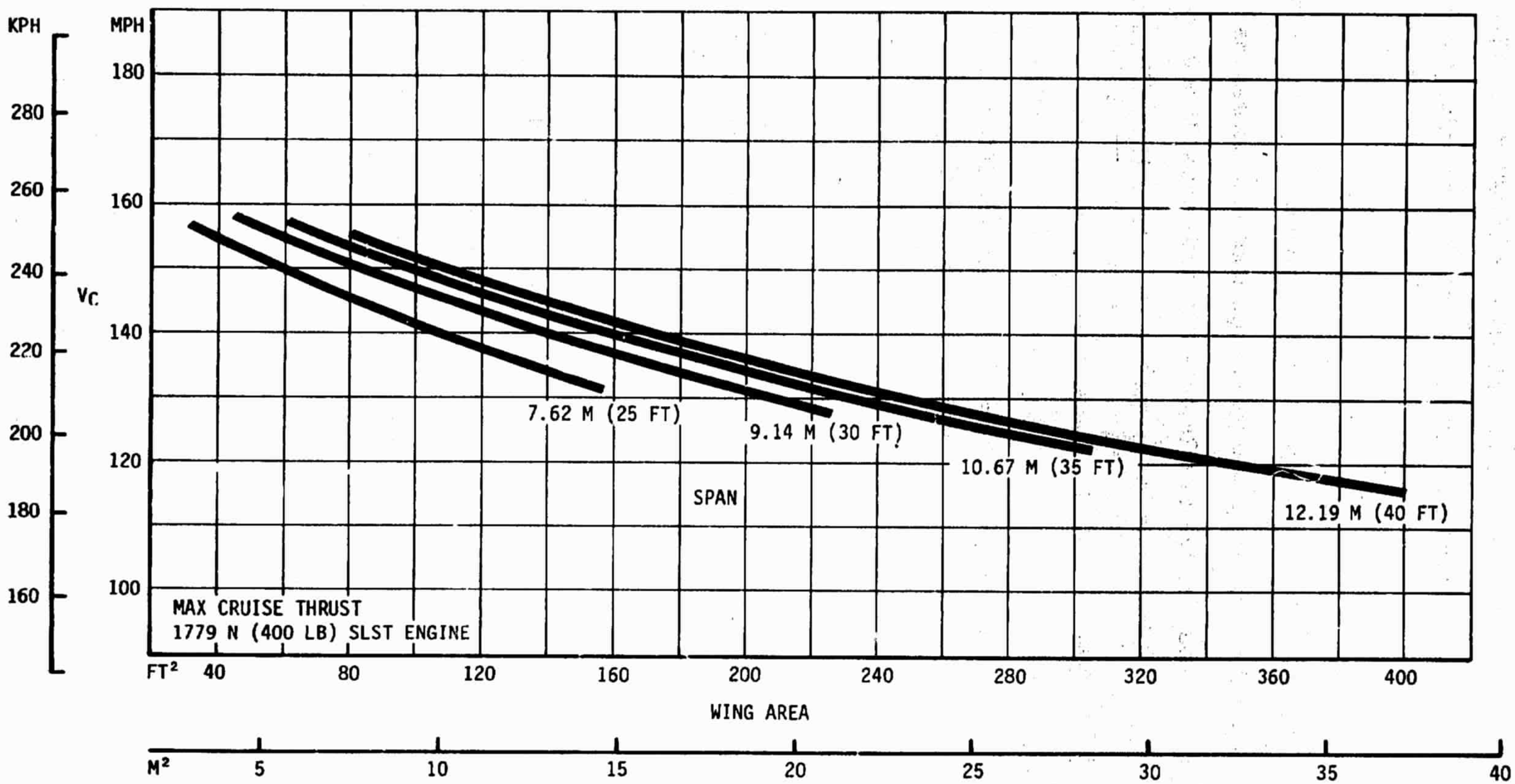

FIGURE 11 - Sea Level Cruise

Figure 12 shows the cruise speeds at 1524 m (5000 ft.). The longer spans are slightly faster at this altitude than at sea level, but the shorter ones are somewhat slower.

The 3048 (10,000 ft.) cruise speeds are shown on Figure 13. The dashed line indicates the specification requirement; note that few configurations exceed it. The 7.6 m (25 ft.) span curve is truncated at both ends; those configurations are incapable of flying at this altitude at cruise thrust. All configurations are slower at this altitude than at 1524 m (5000 ft.).

Figure 14 shows the cruise speeds at 4572 m (15,000 ft.). The 7.6 m (25 ft.) span curve has dropped out completely and the 9.1 m (30 ft.) curve is truncated. Again, all configurations are slower than at 3048 m (10,000 ft.).

The range at 3048 m (10,000 ft.) is shown on Figure 15. Nearly all configurations exceed the specification value of 885 km (550 statute miles). The range shown results from a simple calculation based on fuel quantity, fuel flow, and speed. No allowances were made for taxi, takeoff, climb, descent, landing, or reserves. This estimation technique assumes that the extra fuel burned per mile during climb is made up in descent. While some accuracy is sacrificed by this method, parametric relationships are shown properly.

Figure 16 shows the takeoff ground roll to be dependant only on wing area. This indicates that it is a strict function of stall speed; it is, actually, a function of liftoff speed which is usually a specified margin above stall speed.

The takeoff air distance over a 15 m (50 ft.) obstacle, shown on Figure 17, is the first graph to show a maximum or minimum. It can be assumed that at wing areas less than the minimum point the higher stall and takeoff speeds cause the longer distances, while at areas greater than the minimum, the higher drag increases the distance.

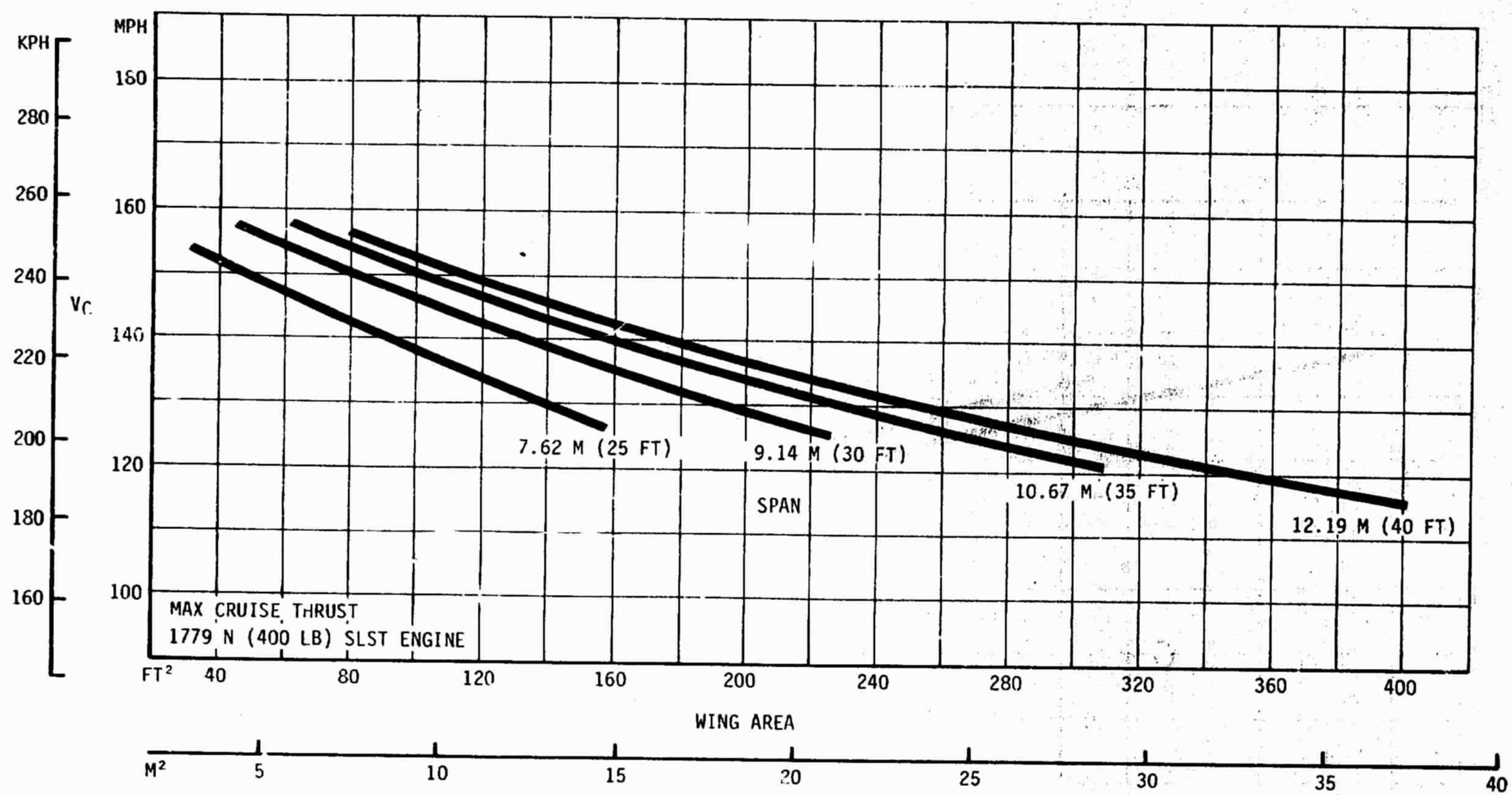

FIGURE 12 - 1524 M (5000 FT) Cruise

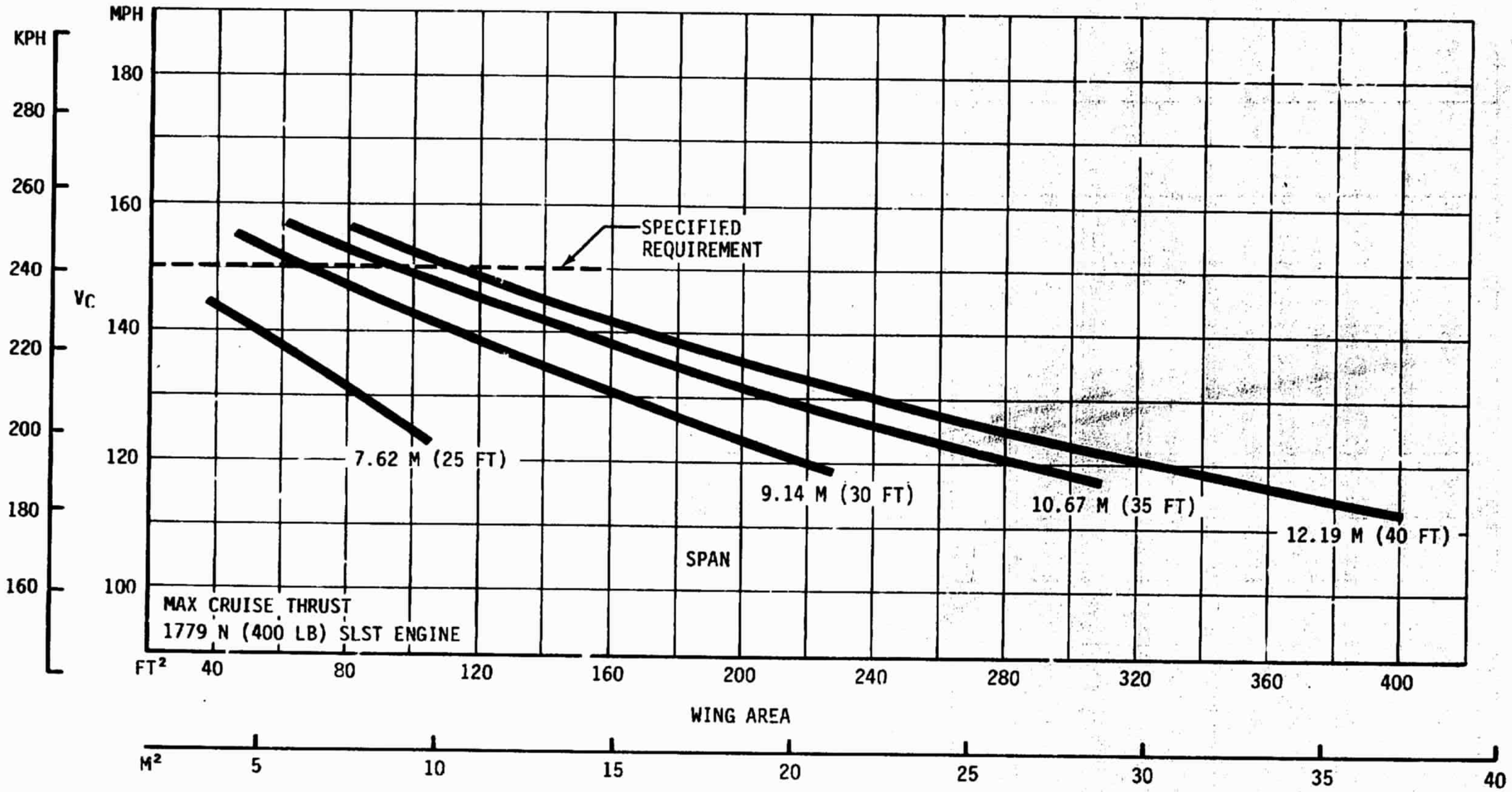

FIGURE 13 - 3048 M (10000 Ft) Cruise

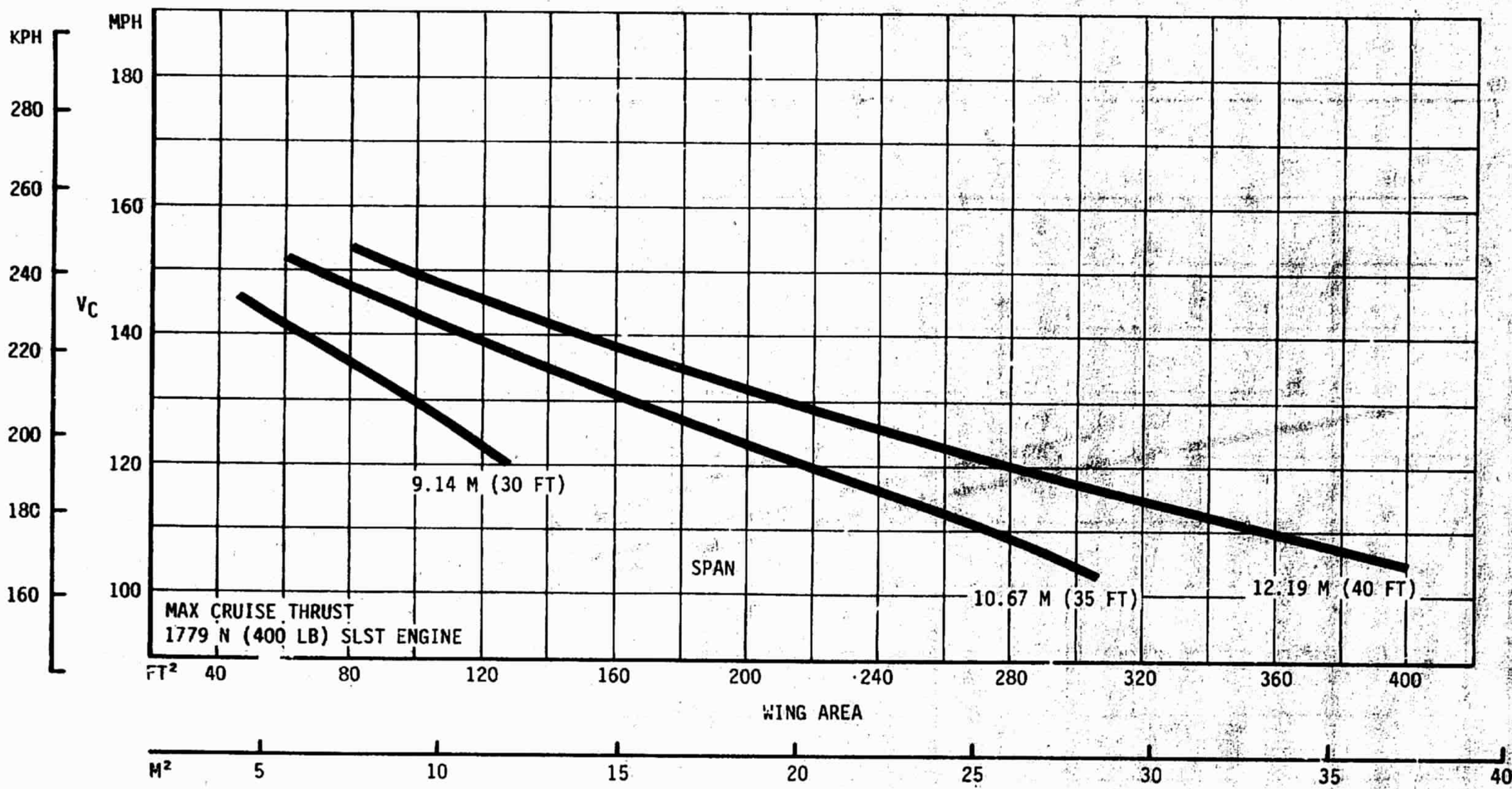

FIGURE 14 - 4572 M (15000 Ft) Cruise

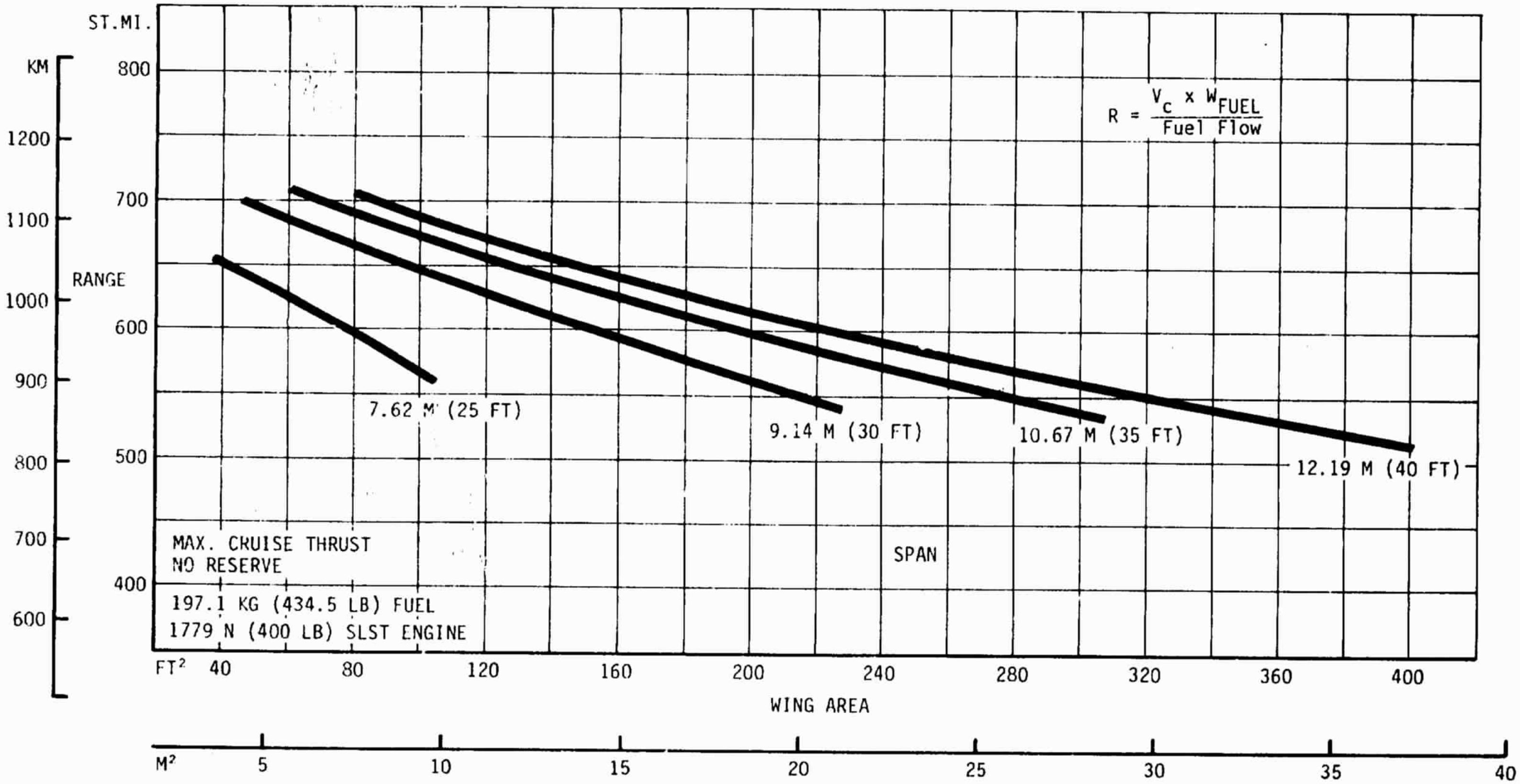

FIGURE 15 - 3048 M (10000 Ft) Range

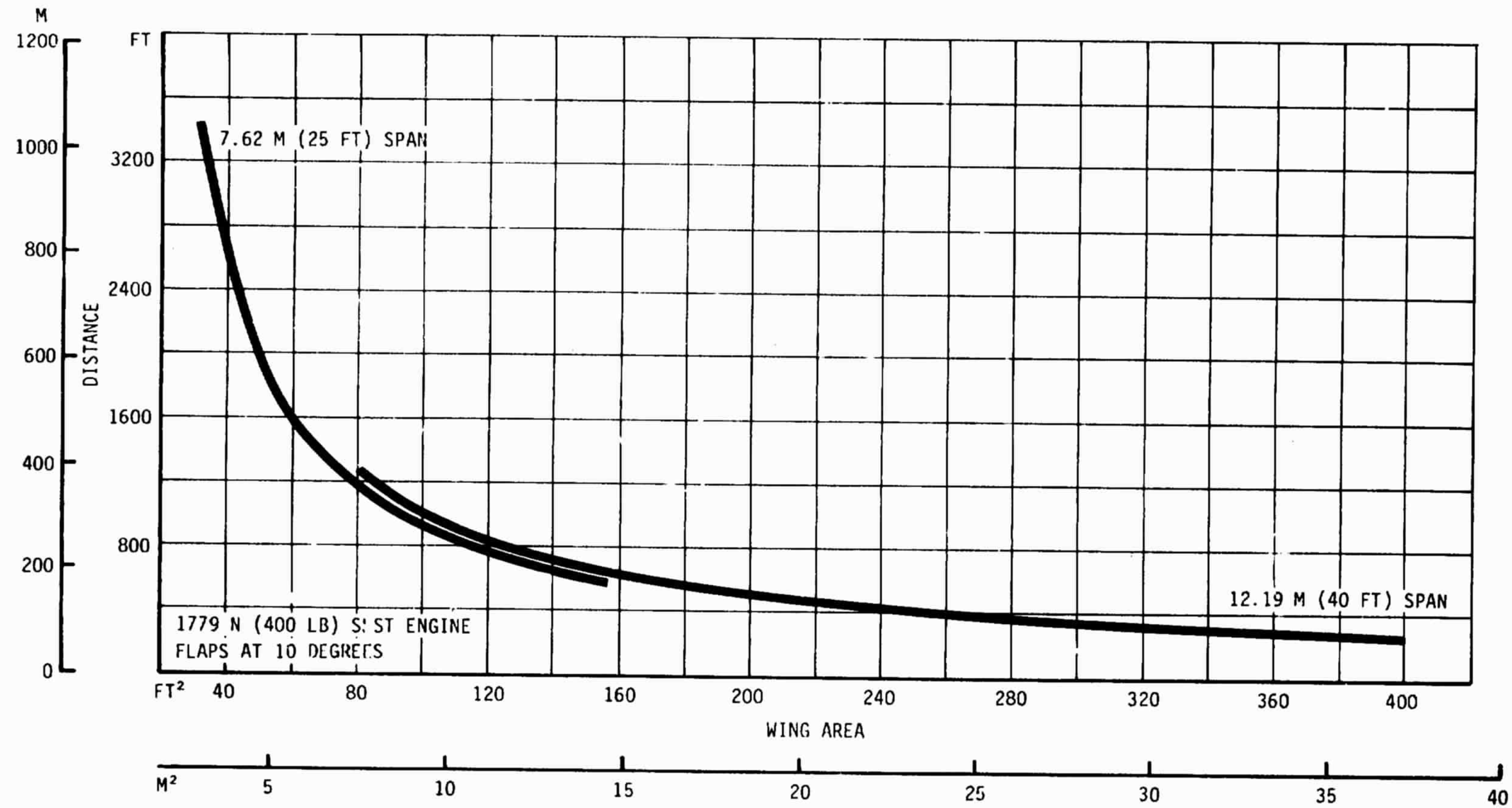

FIGURE 16 - Takeoff Ground Roll

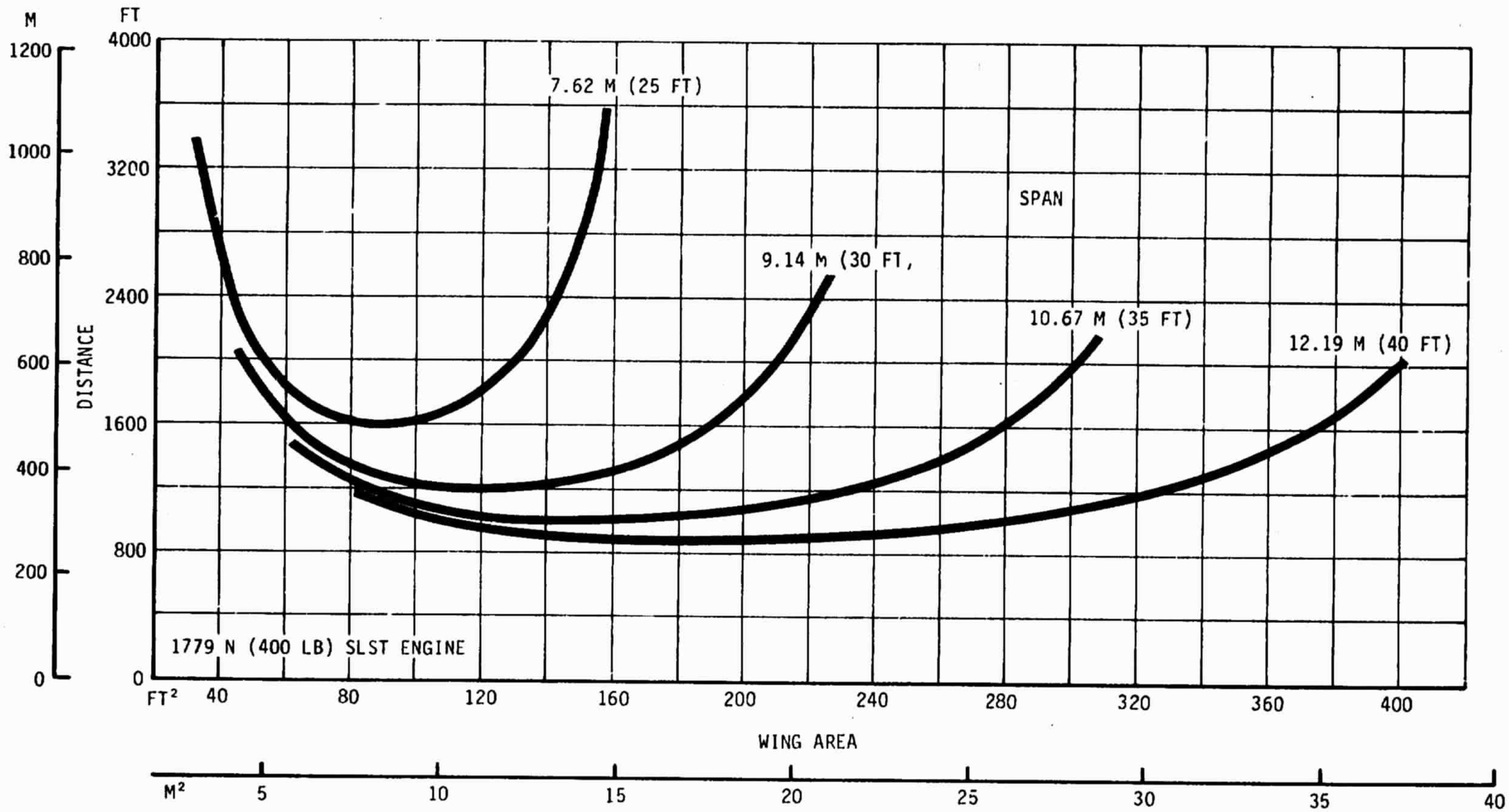

FIGURE 17 - Takeoff Air Distance Over 15 M (50 Ft)

Figure 18 shows the total distance required for takeoff over a 15 m (50 ft.) obstacle. These curves, of course, are simply the sum of the previous two. Note the rather poor distances obtained. No takeoff or landing distances were specified, so the authors defined two levels of performance. Six hundred and ten meters (2000 ft.) total distance was chosen as a required performance level. This was felt to be the maximum distance that would permit safe operation from a typical 914 m (3000 ft.) general aviation runway under all conditions by private pilots. Four hundred and fifty-seven meters (1500 ft.) total distance was assessed as a desirable performance level which would allow competition with comparable piston powered airplanes. The validity of these judgements must remain in question until turbofan airplanes arrive in the marketplace. These two levels of performance are shown by dashed lines on Figure 18. Few configurations meet the 610 m (2000 ft.) criterion, and almost none the 457 m (1500 ft.) level. A check of the data shows that those configurations that require less than 457 m (1500 ft.) have stall speeds with takeoff flaps less than 74 kph (46 mph). The cause of this poor performance is that the assumed engine size is inadequate to meet the desired takeoff performance.

The landing ground roll, Figure 19, shows more dependence on span than the takeoff ground roll. This is felt to be due to differences in the way ground effect affects braking effectiveness. The dashed curves show the effect of deploying spoilers simultaneously with brake application. These are either the lateral control spoilers or similar surfaces installed inboard of them (Reference 4).

Figure 20 shows the air distance required to land over a 15 m (50 ft.) obstacle. Although no spoilers were used for this segment, it is still rather short due to the high drag of the full span Fowler flaps.

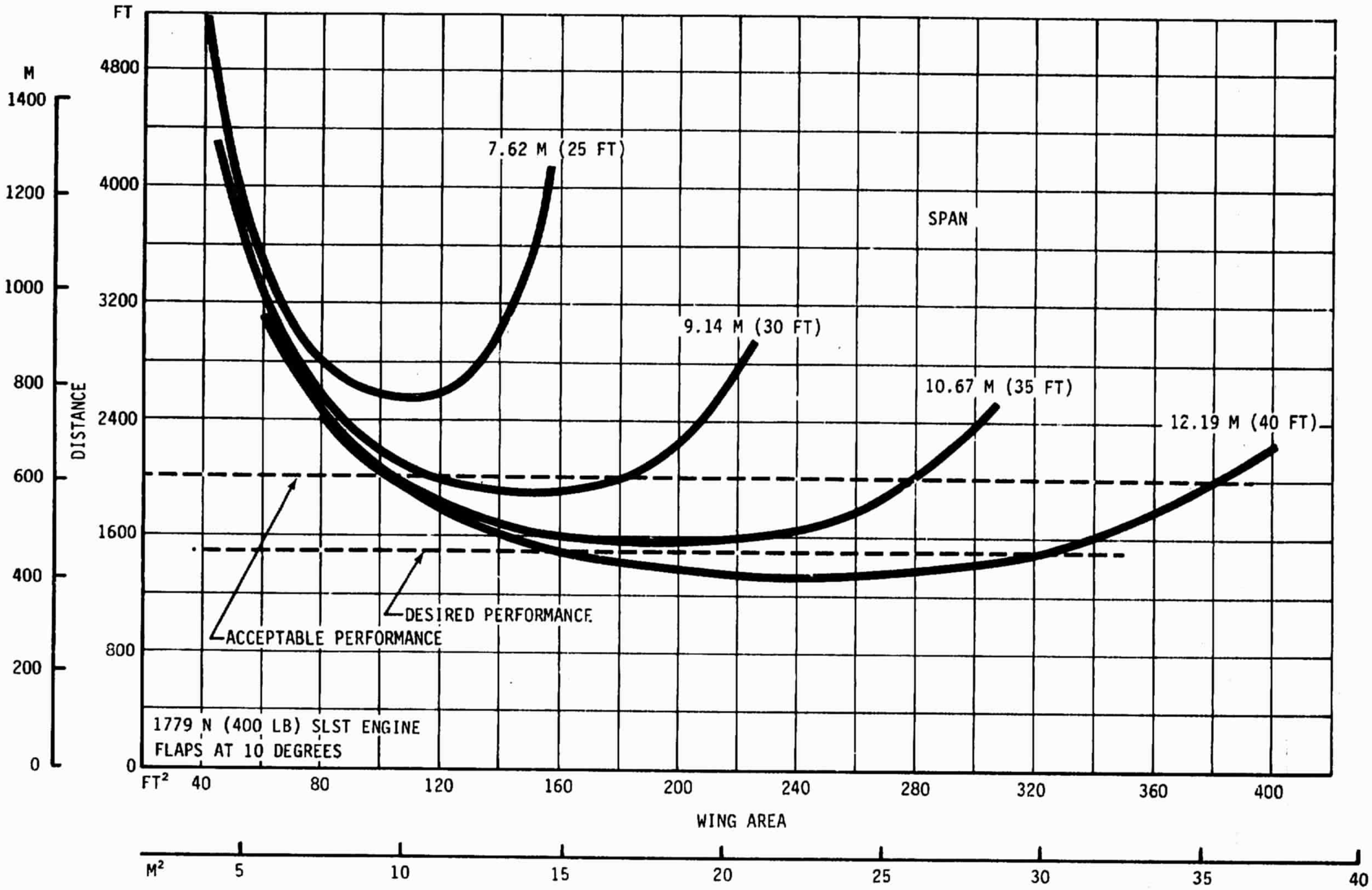

FIGURE 18 - Total Takeoff Distance Over 15 M (50 Ft)

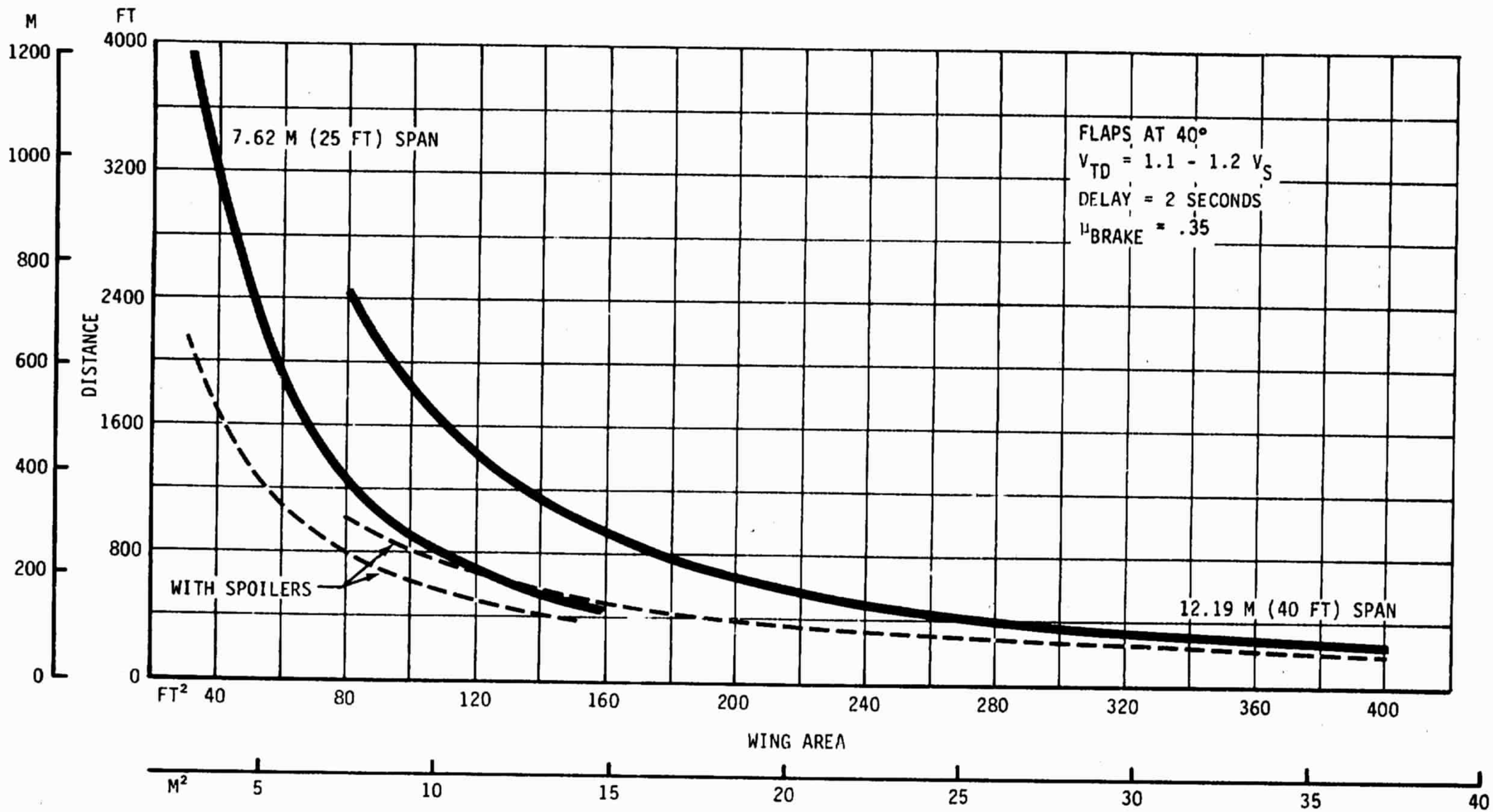

FIGURE 19 - Landing Ground Roll

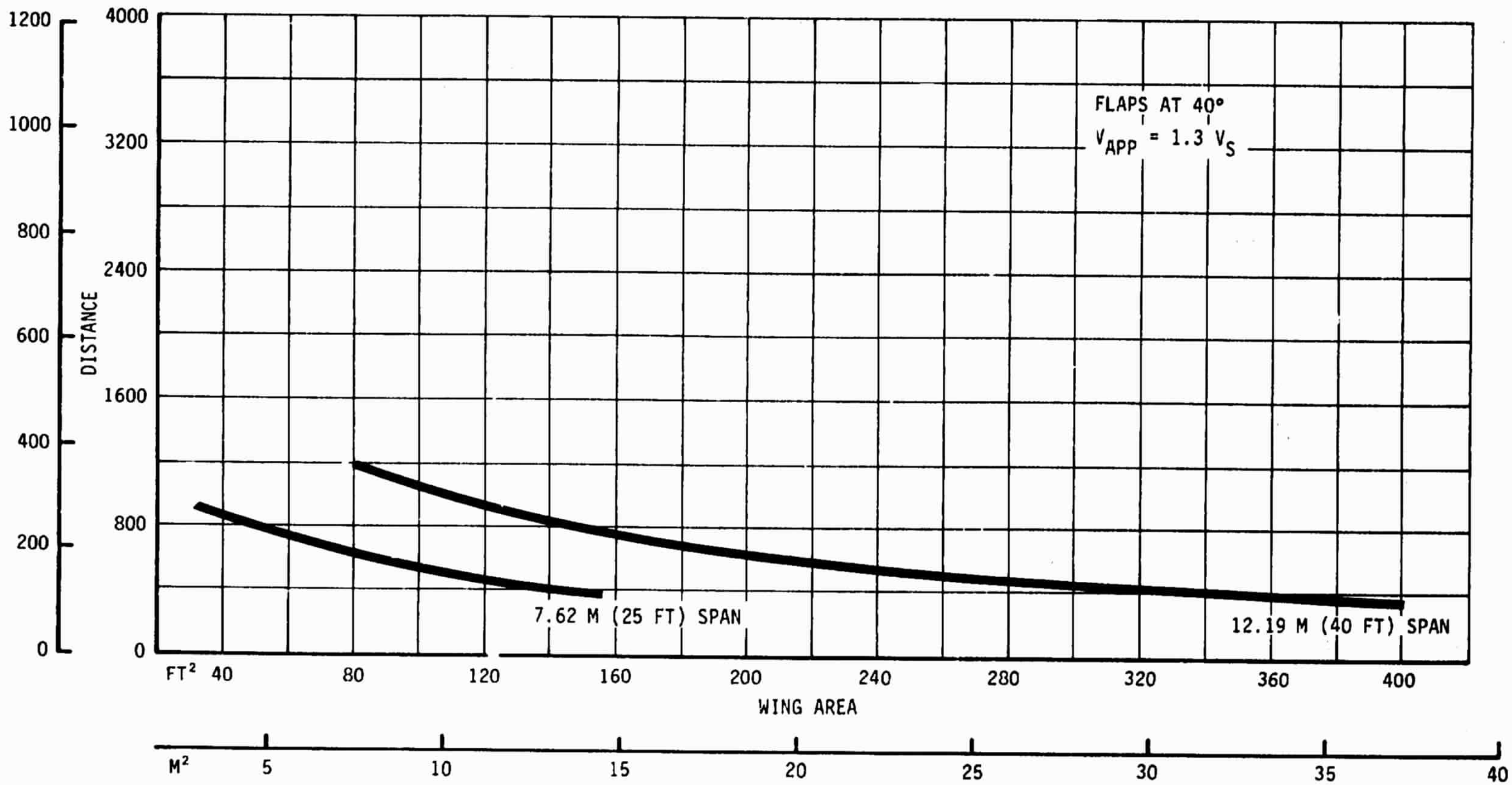

FIGURE 20 - Landing Air Distance Over 15 M (50 Ft)

Figure 21 shows the total landing distance over 15 m (50 ft.). Based on the field performance specified for takeoff, the number of satisfactory configurations is much greater than for takeoff, particularly when spoilers are used.

In view of the poor performance in takeoff and cruise, and the marginal climb performance, it was decided to repeat these calculations with a 2224 N (500 lbs.) sea level static thrust engine.

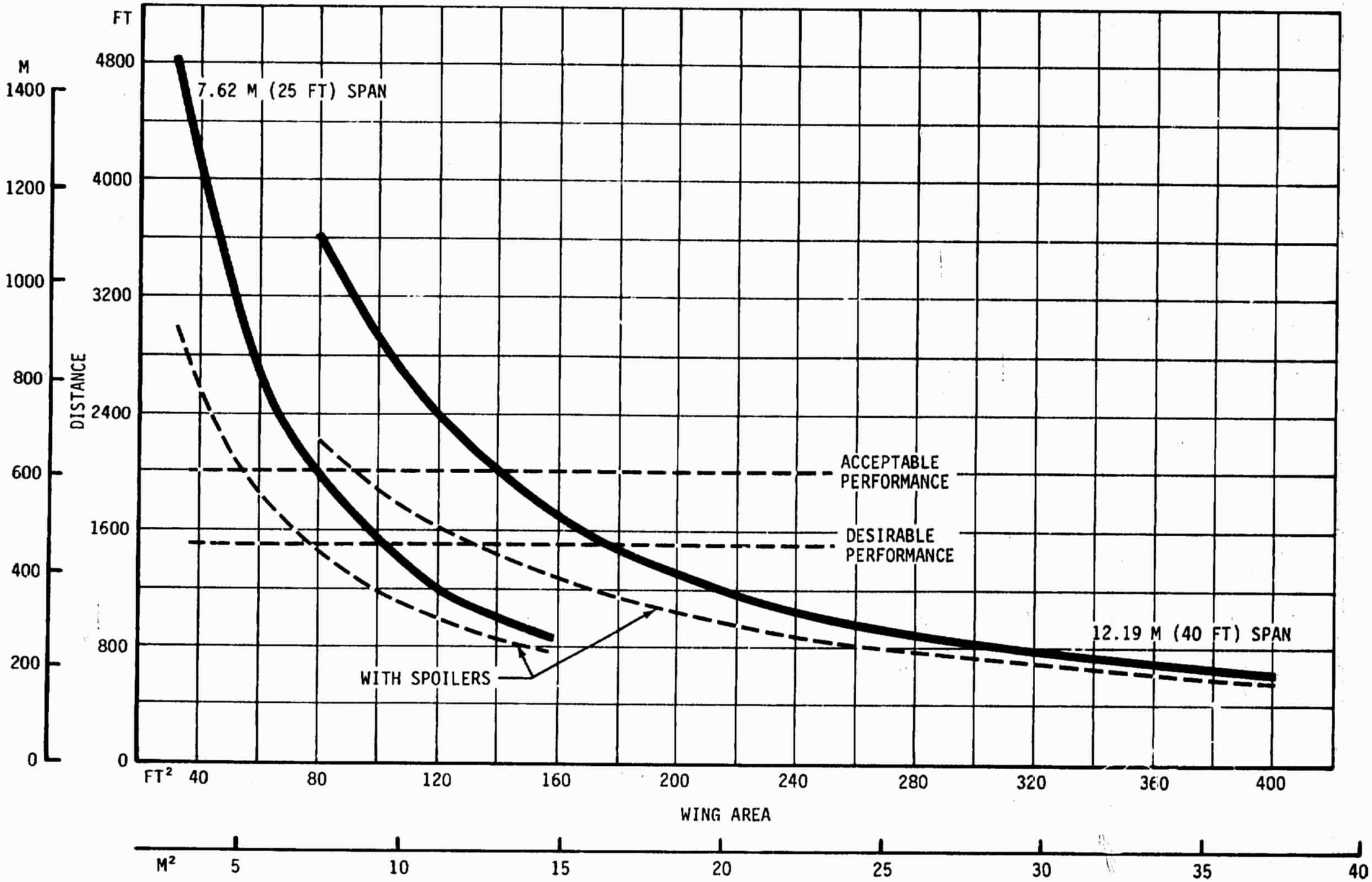

FIGURE 21 - Total Landing Distance Over 15 M (50 Ft)

6.5 Performance - 500 Lb. Engine

Figure 22 shows the revised gross weight. This is the same as Figure 6 except for a 15.9 kg (35 lb.) increase in empty weight due to the heavier engine and the slight increase in wing weight this caused.

The stall speed, Figure 23, is the same except for the small increase due to the heavier weight.

The rate of climb, shown on Figure 24, is considerably higher than the previous configuration, as expected. The FAR Part 23 requirement now exerts little restriction. The service ceilings, Figure 25, range from good to outstanding with this power plant.

Figure 26 shows the new top speed. The variation with span and area is similar to the previous one, except that the speed levels are higher. Similarly, the cruise speed at 3048 m (10,000 ft.), Figure 27, is much improved with most of the configurations now falling above the requirement. The range has changed in the opposite direction, Figure 28, and all configurations fall short of the requirement. This is not a significant problem, however, since the fuel capacity of the selected configuration may simply be increased to meet the requirement (with a corresponding increase in gross weight).

The takeoff ground roll, Figure 29, is similar to the previous one, except for the shorter distances. The air distance, Figure 30, is also reduced and the minima are less pronounced. The total takeoff distance, Figure 31, is greatly reduced and most configurations now meet the 457 m (1500 ft.) criterion.

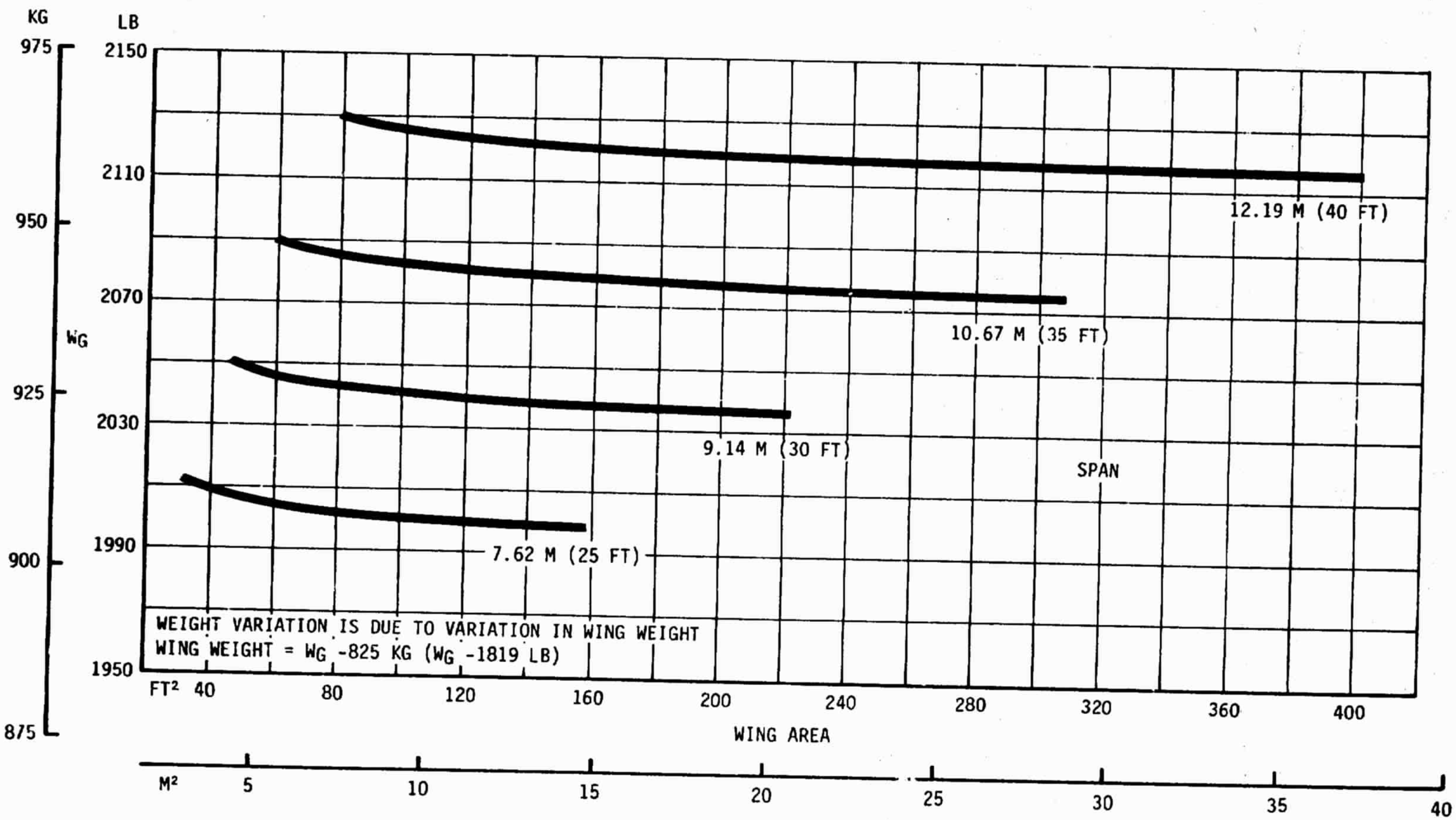

FIGURE 22 - Gross Weight

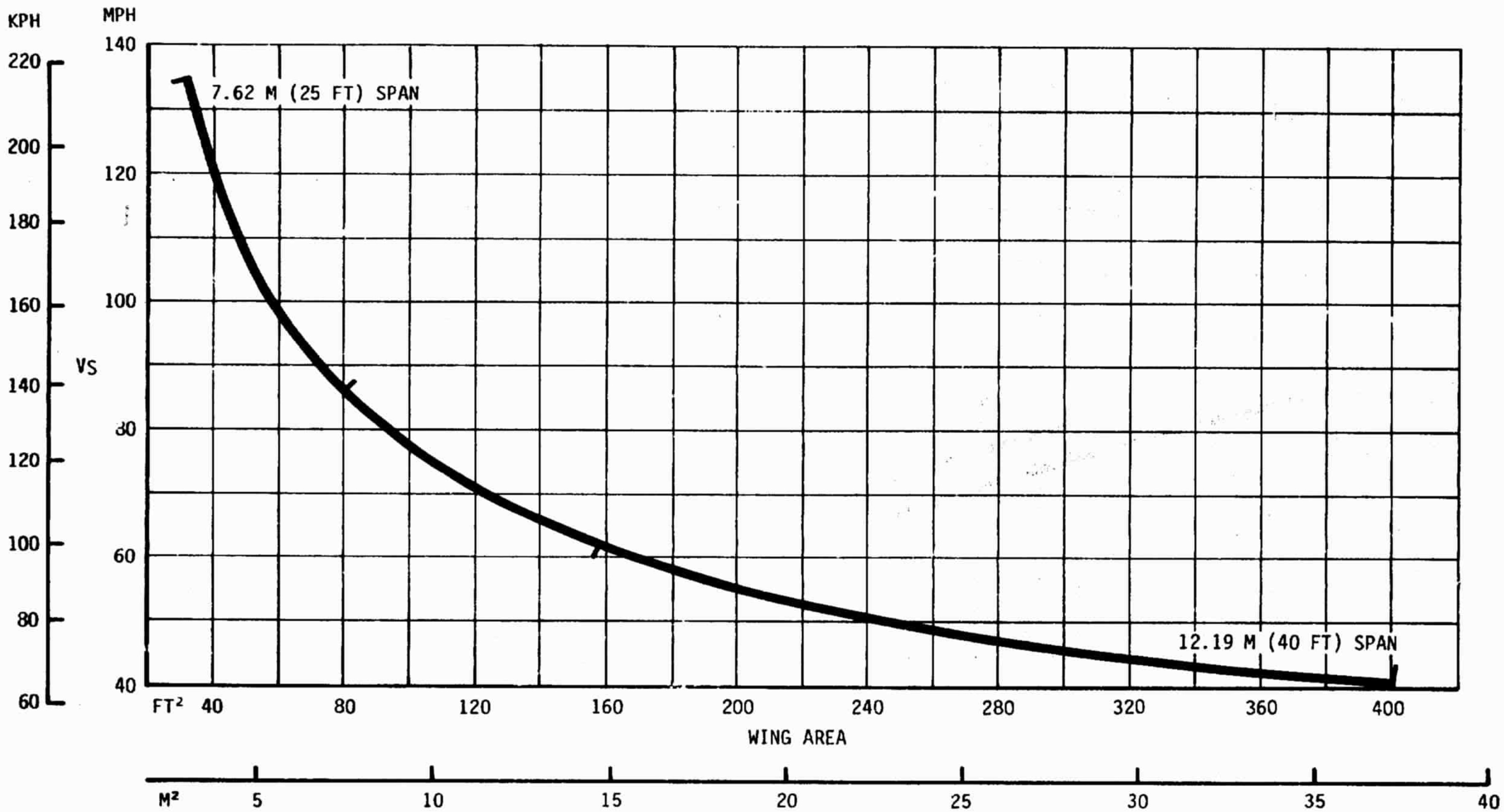

FIGURE 23 - Flaps Up Stall Speed

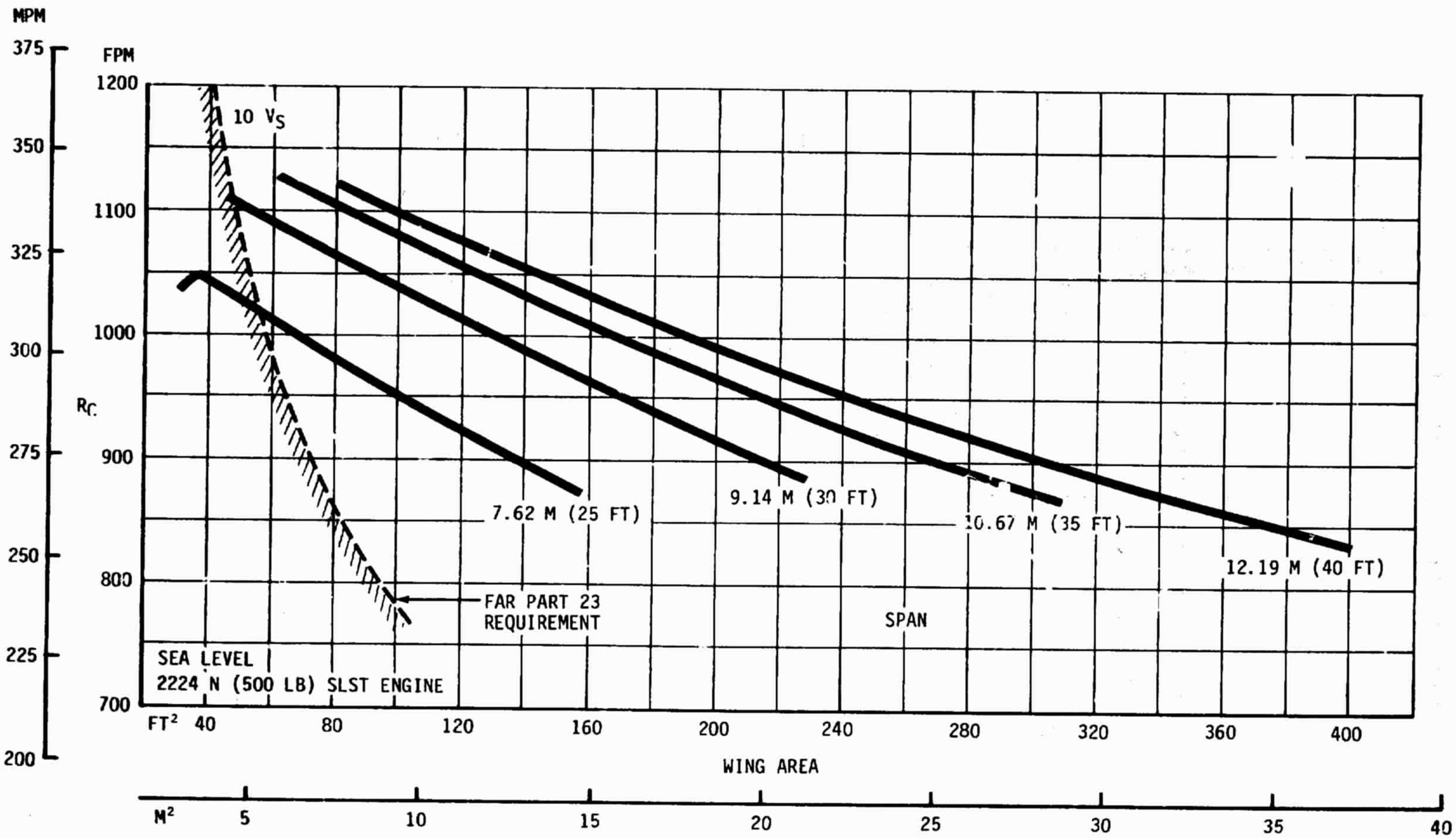

FIGURE 24 - Rate of Climb

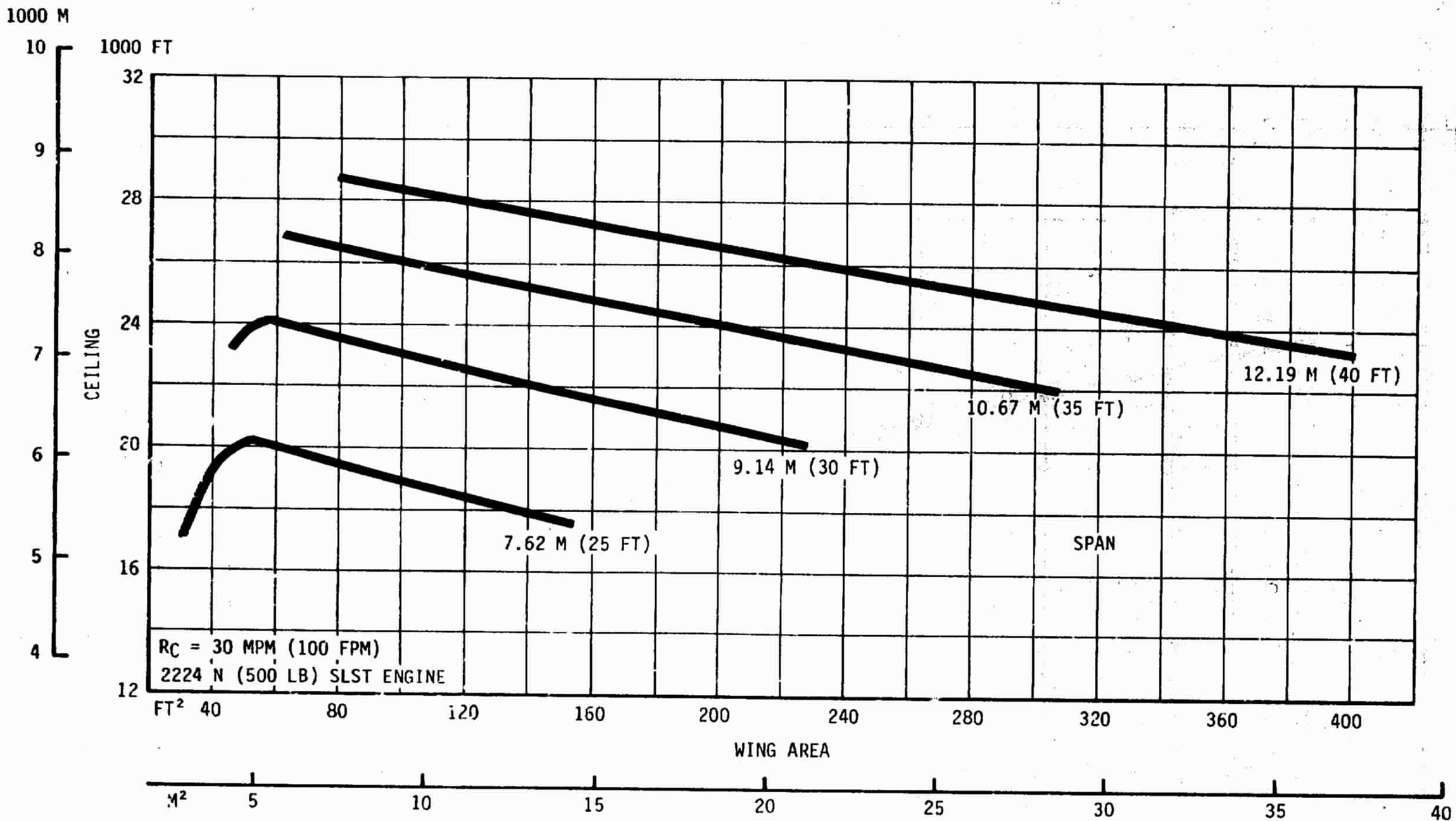

FIGURE 25 - Service Ceiling

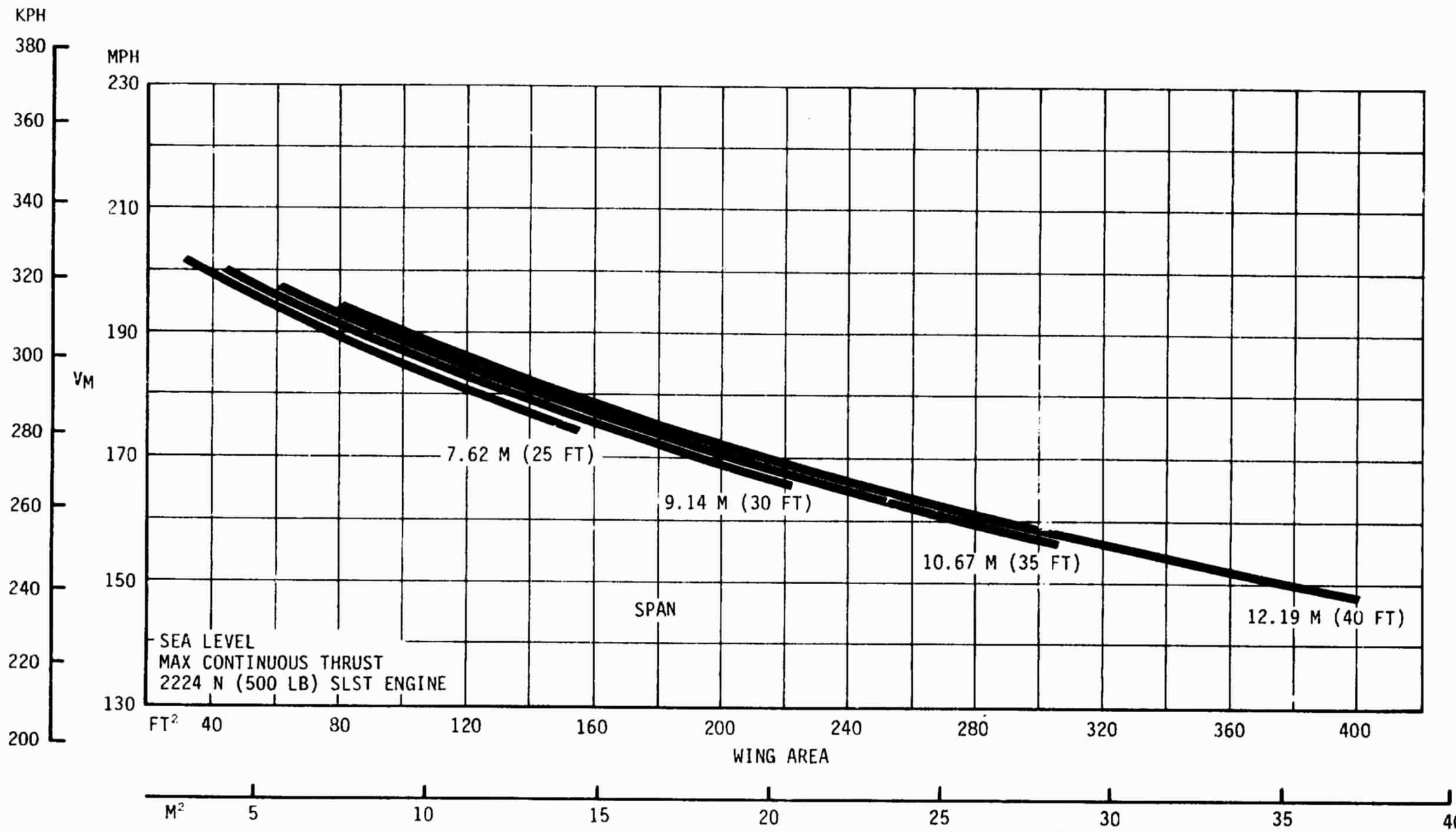

FIGURE 26 - Top Speed

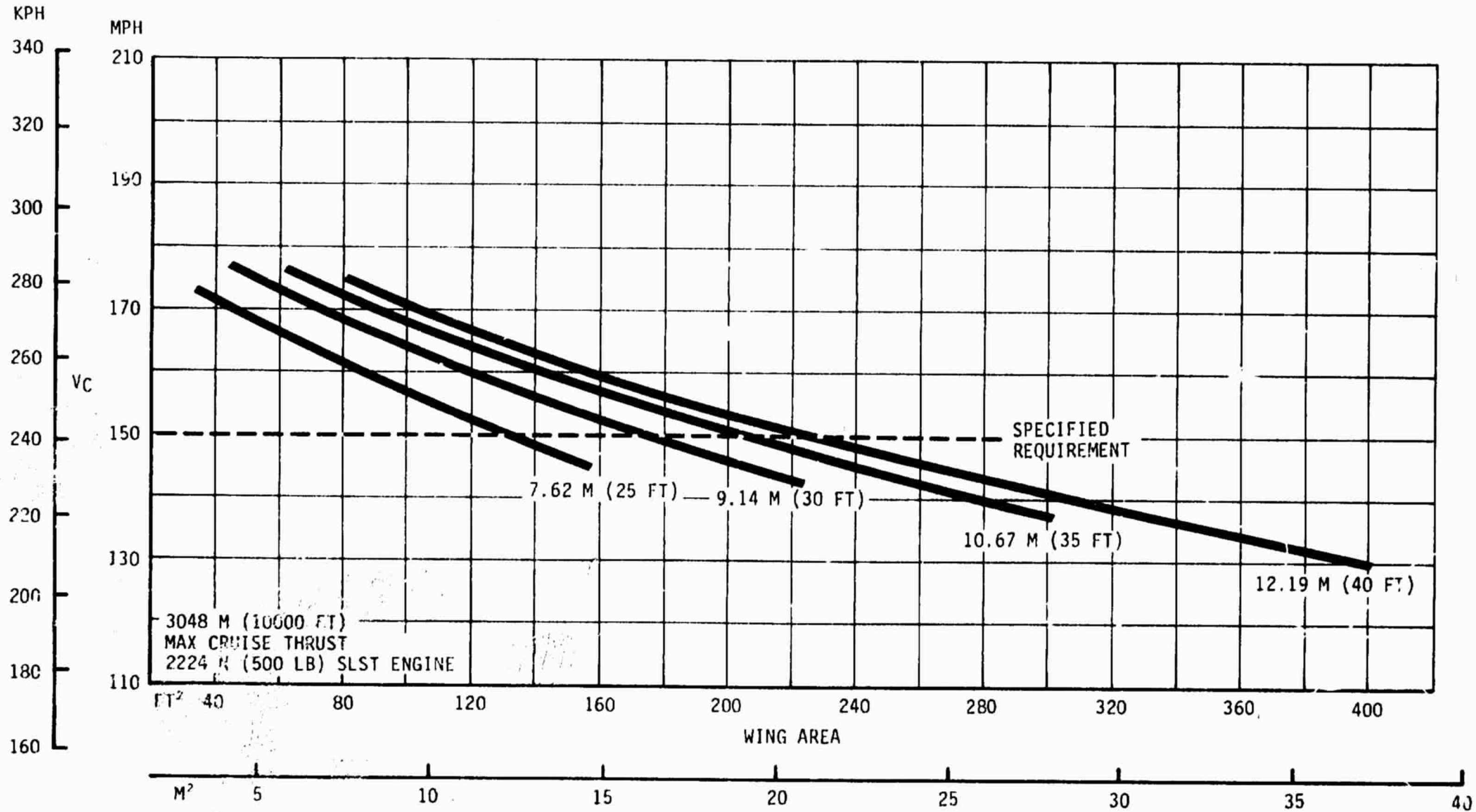

FIGURE 27 - Cruise Speed

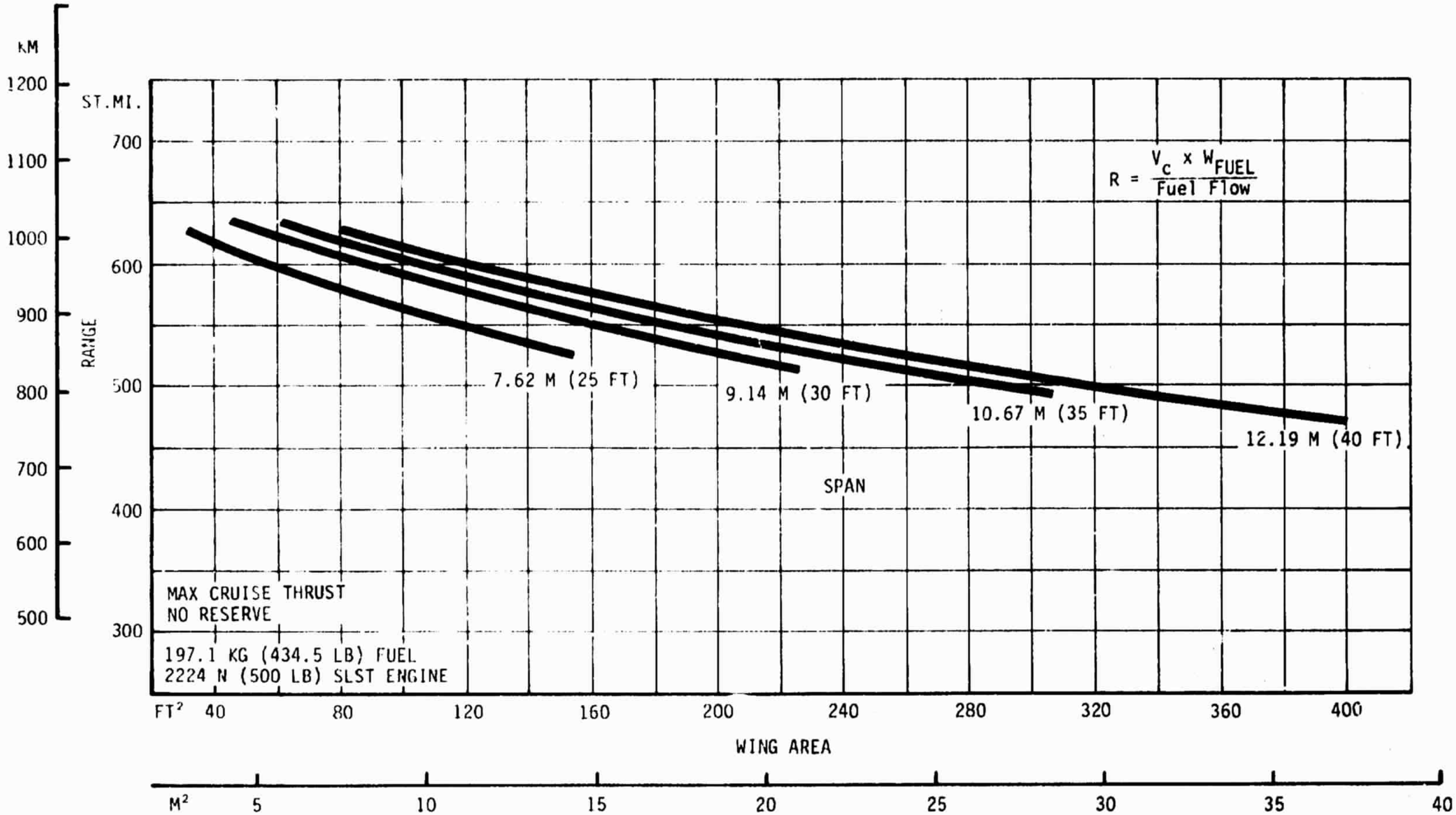

FIGURE 28 - 3048 M (10000 Ft) Range

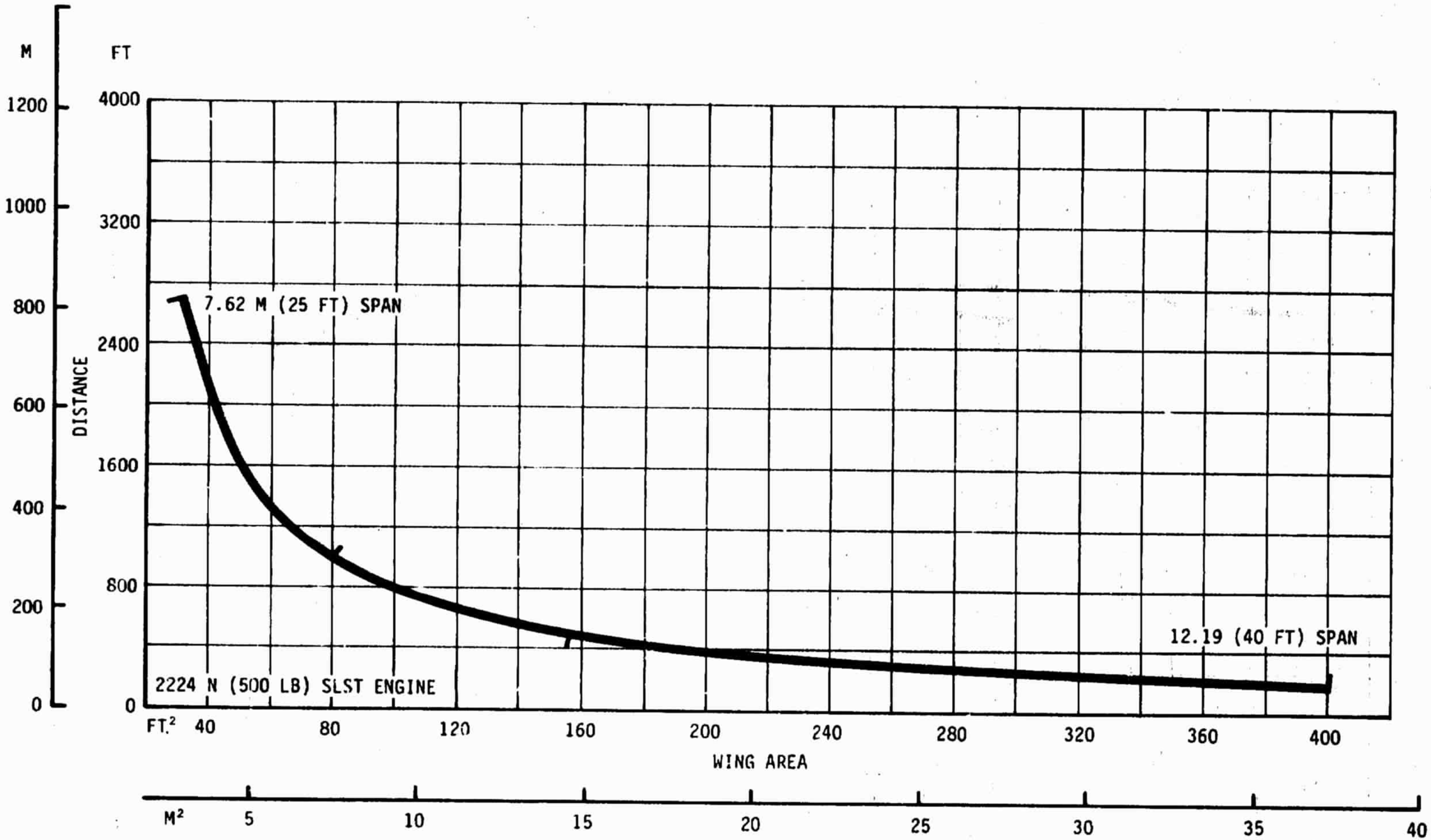

FIGURE 29 - Takeoff Ground Roll

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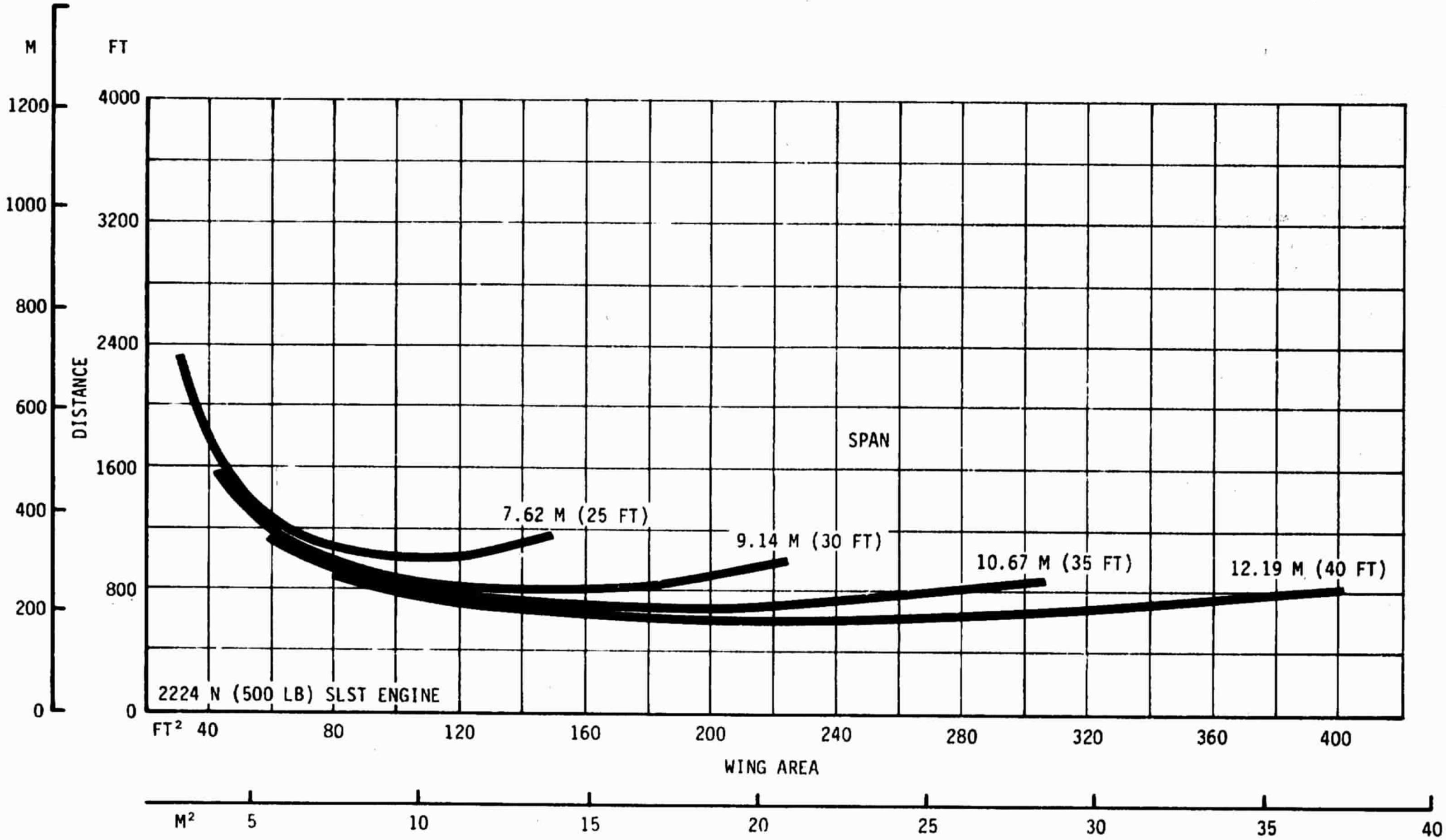


FIGURE 30 - Takeoff Air Distance Over 15 M (50 Ft)

The engine change would not affect the landing distance, except for the negligible increase due to the higher gross weight; thus, it was not recalculated.

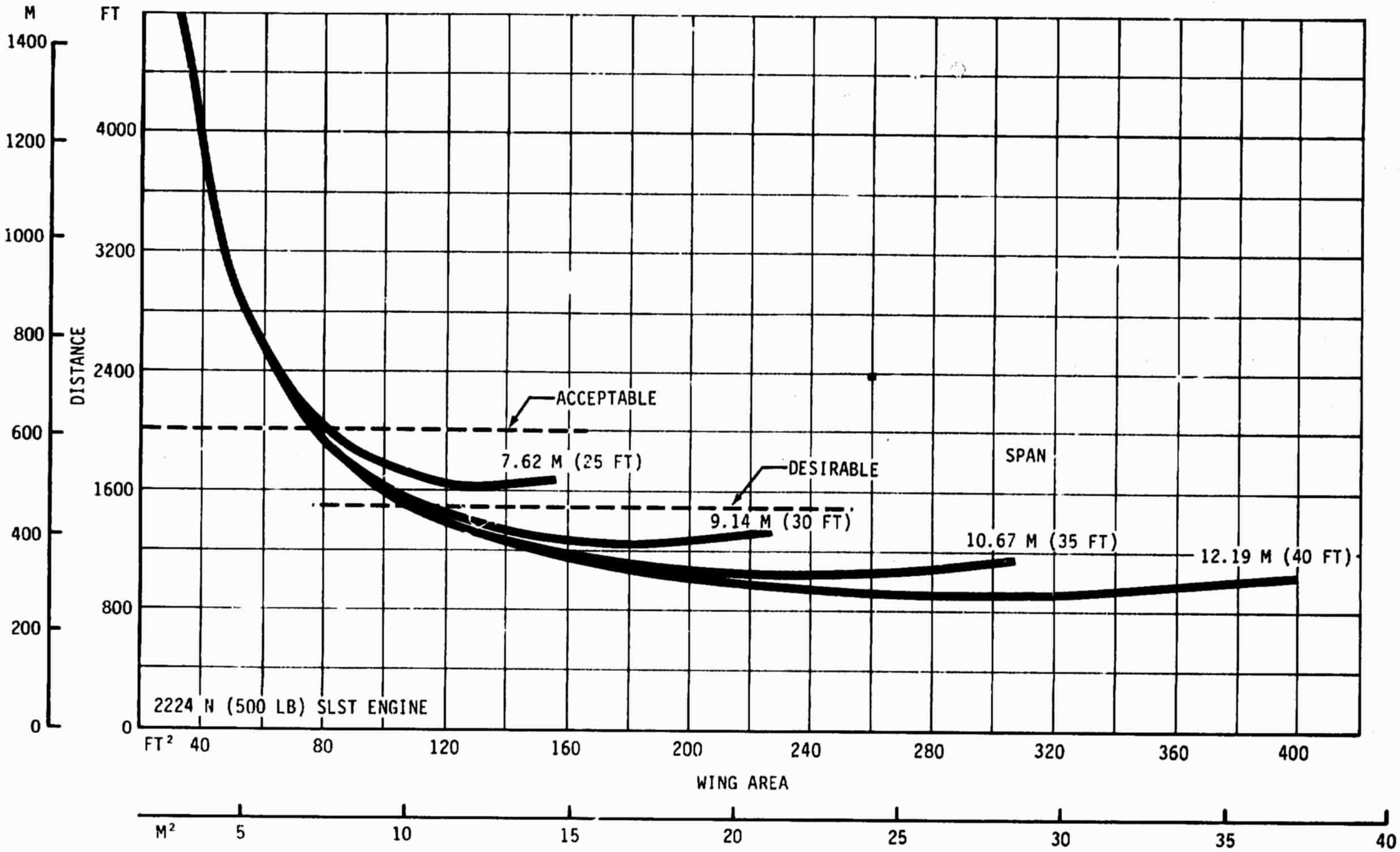


FIGURE 31 - Total Takeoff Distance Over 15 M (50 Ft)

6.6 Configuration Selection

The selection process should maximize cruise speed, range, rate of climb, and service ceiling, and minimize stall speed and takeoff and landing distances. Since cost is a direct function of airframe and weight, gross weight should be minimized. In terms of this study, span should be minimized to minimize weight and cost. Area should be minimized to maximize speed and climb. Area and span should be maximized to minimize takeoff and landing distances. In order to reconcile these conflicting effects, the requirements were plotted on one graph to define the area in which freedom of selection existed. The rate of climb graph was chosen for this purpose.

Figure 32 shows this plot for the 1779 N (400 lb.) engine. The stall speed (V_S) curve is the same as discussed earlier; configurations above and to the right of it are acceptable. The cruise speed (V_C) curve is a cross plot of the cruise speed requirement; configurations above and to the left of it are acceptable. However, none of the configurations remaining above both these curves meet the desirable 1500 ft. takeoff and landing criterion, although they do require less than 610 m (2000 ft.).

With the 2224 N (500 lb.) engine, Figure 33, a much larger area is available to select from. The rate of climb (R/C) curve, on the left side of the plot, is not critical. The acceptable area is above the cruise speed (V_C) curve, below the 457 m (1500 ft.) landing distance (LDG 1500) curve, and to the right of the 457 m (1500 ft.) takeoff distance (T.O. 1500) curve. Within this area, as noted above, span should be minimized to minimize weight, and wing area should be minimized to maximize speed and rate of climb. The gross weights and ranges for each configuration represented in Figure 33 may

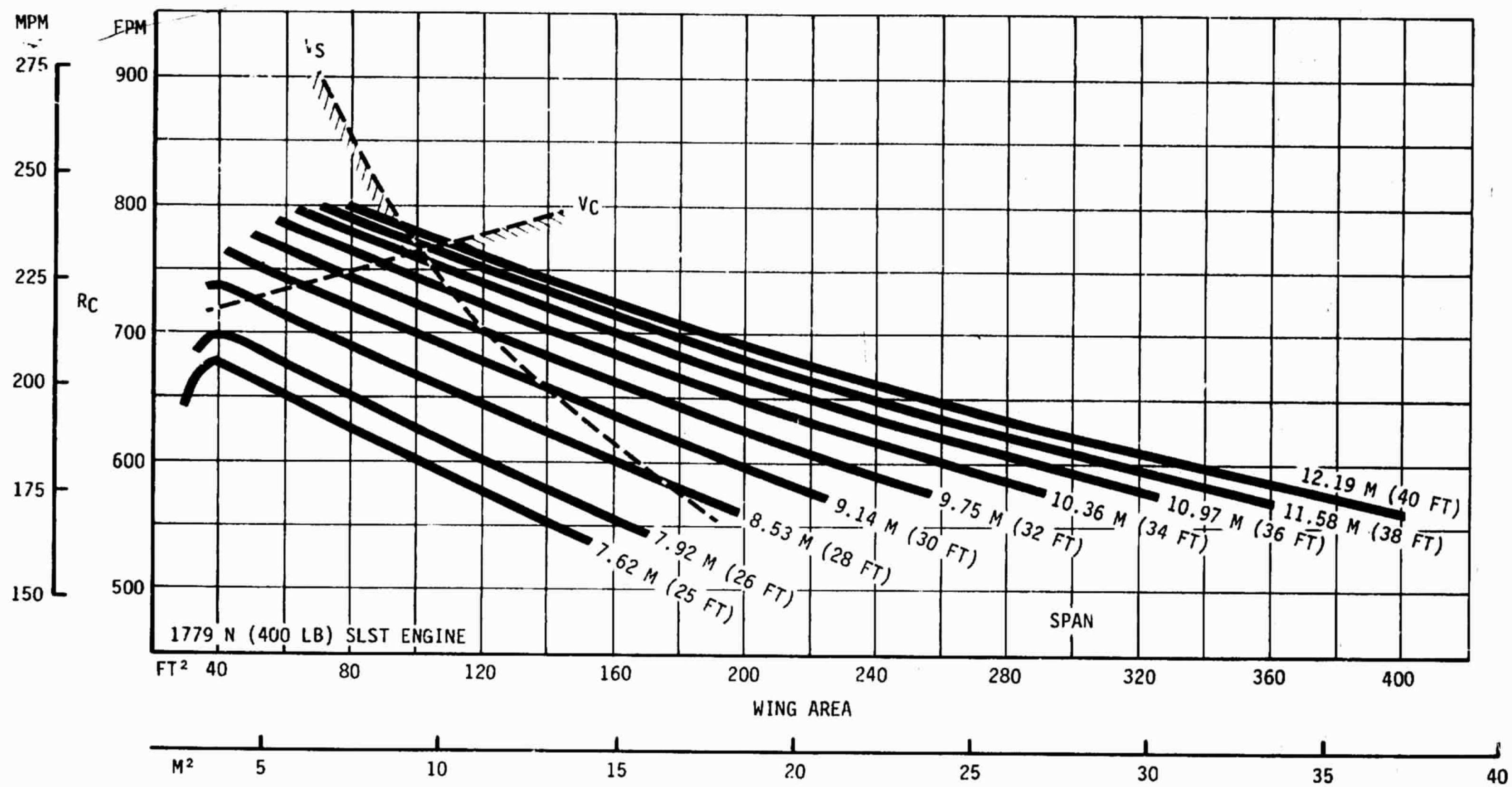


FIGURE 32 - Configuration Selection Chart

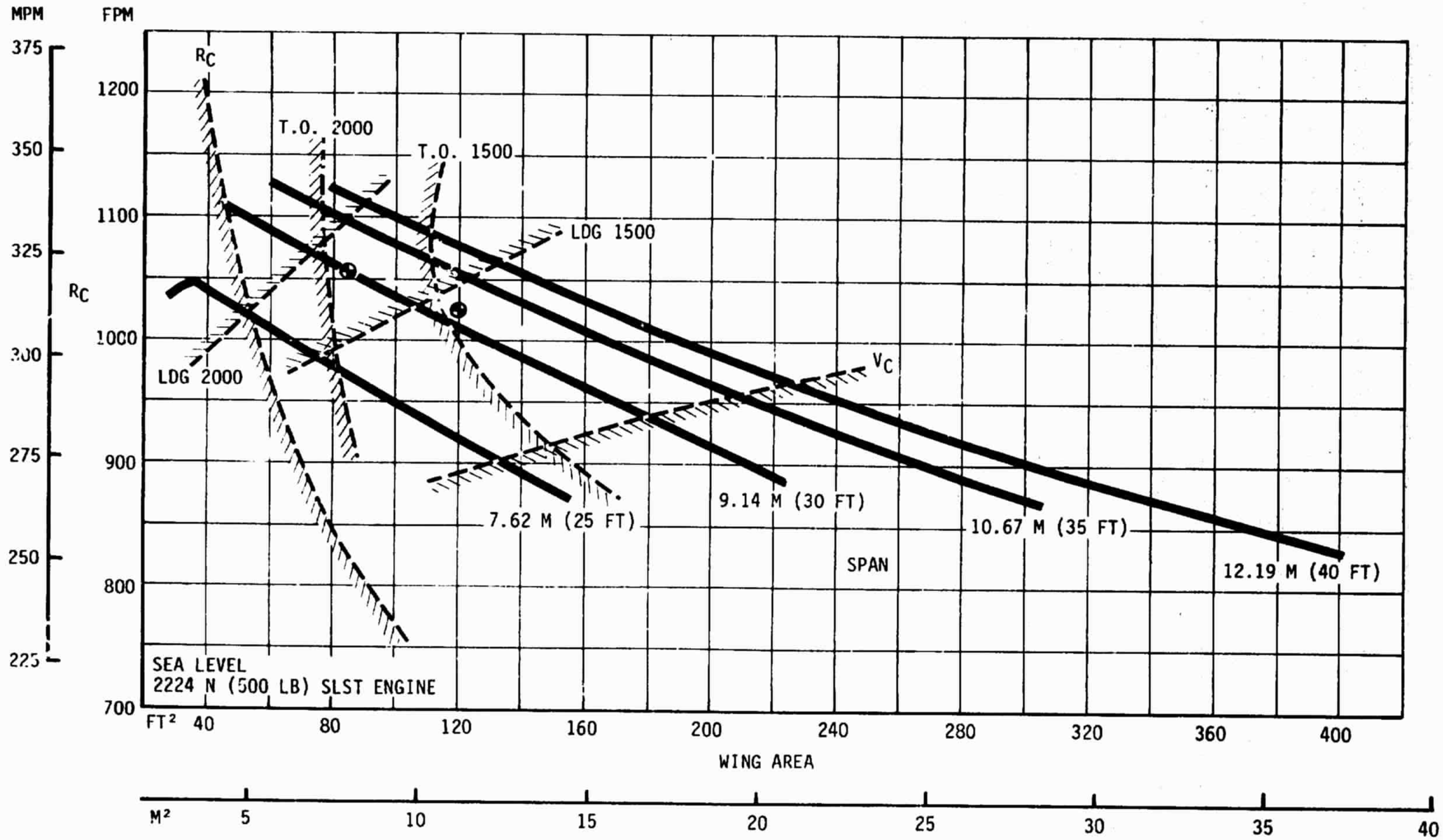


FIGURE 33 - Configuration Selection Chart

be found from Figures 22 and 28, respectively. Some margin should be maintained from the takeoff and landing distance boundaries, because the gross weight will be increased to provide enough fuel for the required range. Note that as long as the rate of climb and takeoff and landing requirements are met, stall speed plays no part in the selection, if the full flap stall speed is less than 113 kph (70 mph) (FAR 23.49).

The configuration chosen from this chart has a wing span of 9.75 m (32 ft.) and a wing area of 11.15 m^2 (120 $ft.^2$). If the takeoff and landing requirement were relaxed to 610 m (2000 ft.), an alternate configuration with a smaller wing of 9.1 m (30 ft.) span and 7.9 m^2 (85 $ft.^2$) area would be acceptable. Table 2 lists the performance of these airplanes from the preceding figures. Figure 34 shows the selected configuration.

The selected configuration was further modified by increasing the tail size to be compatible with the larger wing. Since it was not necessary to increase the fuel supply to meet the range requirement, this final change brought the final gross weight to 935 kg (2061.3 lbs.), as shown in the revised weight breakdown given in Table 3. The weight and balance calculations were repeated. The leading edge of the wing was maintained at the same location as the previous configuration. The most forward c.g. was located at 11.48% MAC, the most aft c.g. was at 31.84% MAC. These were considered satisfactory, pending the stability and control calculations.

TABLE 2

PERFORMANCE SUMMARY

	Selected Configuration	Alternate Configuration
Wing Span, m (ft)	9.75 (32)	9.1 (30)
Wing Area, m^2 (ft^2)	11.15 (120)	7.9 (85)
Aspect Ratio	8.53	10.59
Gross Weight, kg (lb)	933 (2057)	927.1 (2044)
Stall Speed, kph (mph) flaps up	114 (71)	135 (84)
Rate of Climb, mpm (fpm)	313 (1026)	321 (1054)
Service Ceiling, m (ft)	7254 (23800)	7132 (23400)
Top Speed, kph (mph)	296 (184)	307 (191)
3048 m (10000 ft) Cruise Speed, kph (mph)	261 (162)	270 (168)
Range, km (mi) (197.1 kg (434.5 lb) fuel)	937 (582)	969 (602)
Takeoff Ground Roll, m (ft)	197 (645)	276 (907)
Takeoff Air Distance, m (ft)	246 (807)	291 (955)
Total Takeoff Distance, m (ft) to 50 ft.	443 (1452)	567 (1862)
Landing Ground Roll, m (ft)	186 (610)	258 (847)
Landing Air Distance, m (ft)	202 (663)	229 (752)
Total Landing Distance, m (ft) from 50 ft.	388 (1273)	487 (1599)

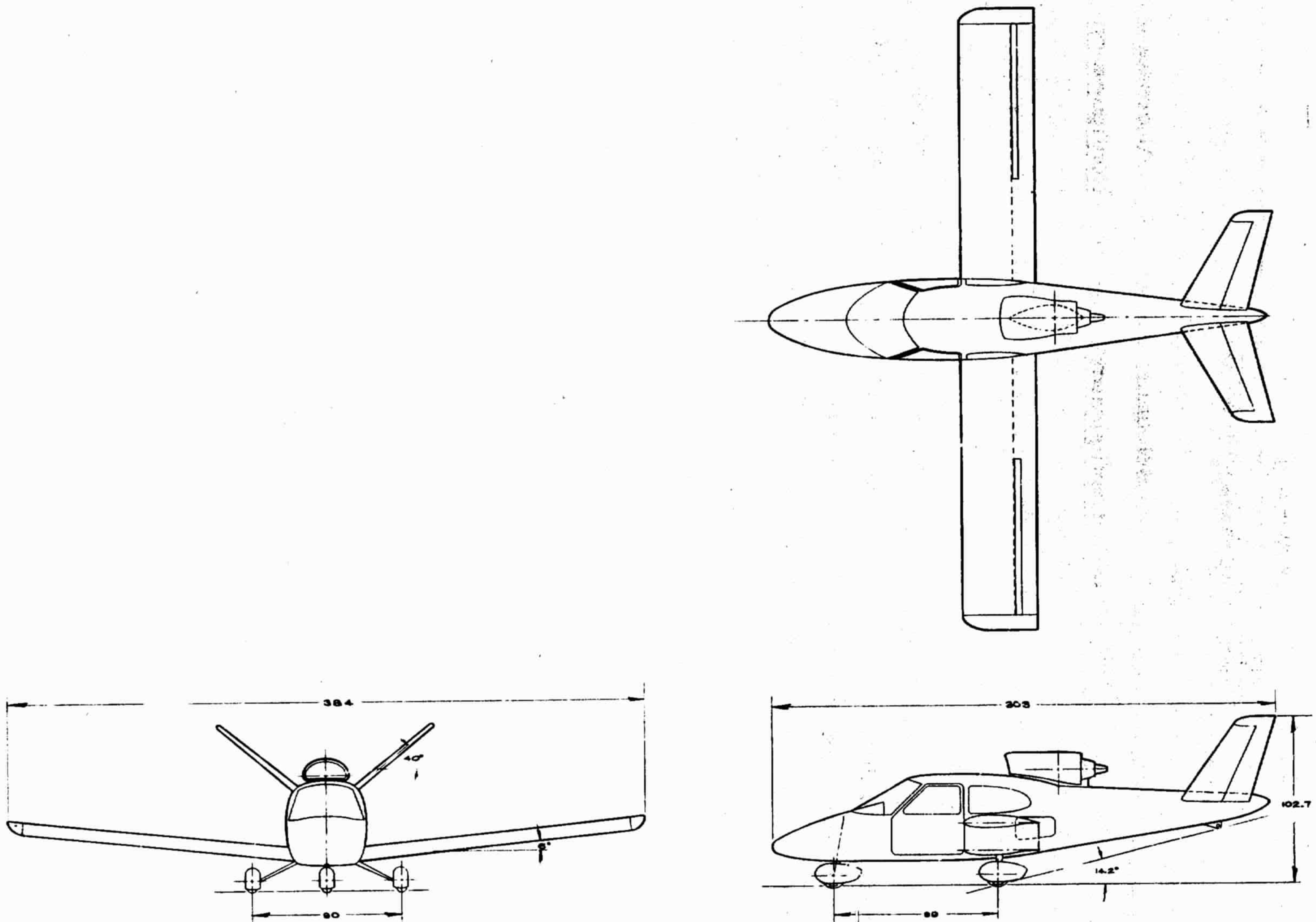


FIGURE 34 - Selected Configuration
PD1502A General Arrangement

TABLE 3

PD1502A WEIGHT SUMMARY

(Weight in kg (lb))

GROUP	WEIGHT
Wing	108.0 (238.0)
Tail Surfaces	21.1 (46.5)
Fuselage	83.9 (185.0)
Landing Gear	49.0 (108.0)
Controls	20.4 (45.0)
Nacelle	10.0 (21.9)
Propulsion	72.1 (159.0)
Instruments	10.8 (23.7)
Avionics	18.1 (40.0)
Electrical	22.7 (50.0)
Furnishings	45.4 (100.0)
Dry, Bare Empty Weight	461.4 (1017.1)
Paint	3.6 (8.0)
Unusable Fuel	2.7 (6.0)
Licensed Empty Weight	467.7 (1031.1)
Payload (Design)	272.2 (600.0)
Maximum Fuel	197.1 (434.5)
Gross Weight	937 (2065.6)

6.7 Stability and Control

The longitudinal static stability and the longitudinal control characteristics were calculated by the methods of Reference 5. There is the possibility that the flow induced by the engine exhaust might affect the tail, thus causing trim changes with power. It is felt that with the low exhaust velocity of the high bypass ratio engine this would not be a serious problem. This is obviously a matter of judgement and the question would be resolved by wind tunnel tests. The equivalent effective area of the vee tail for stability calculations is taken from Reference 6 as

$$S_H = S_{vee} \cos^2 \Gamma$$

since $S_{vee} = 3.58 \text{ m}^2$ (38.52 ft^2), and

$$\Gamma = 40^\circ$$

$$S_H = 2.10 \text{ m}^2 \ (22.60 \text{ ft}^2)$$

For longitudinal control, the effective area is the projected area:

$$S_H = S_{vee} \cos \Gamma = 2.74 \text{ m}^2 \ (29.51 \text{ ft}^2)$$

<u>Stability.</u> - The stability of the airplane is the sum of contributions of the tail, the fuselage, the powernlant, and the c.g. position. Since the powerplant of this airplane is close the the c.g., its effect on stability is assumed to be negligible.

The tail contribution, stick free, is given by

$$\left.\frac{dC_M}{dC_L}\right|_{tail} = -\frac{a_t}{a_w}\,\overline{V}\,\eta_t\left(1-\frac{d\epsilon}{d\alpha}\right)\left(1-\tau\,\frac{C_{h\alpha}}{C_{h\delta}}\right)$$

$$a_t = .058$$

$$a_w = .1078$$

$$\overline{V} = \frac{S_H}{S} \times \frac{\ell_t}{MAC} = \frac{22.60}{120} \times \frac{144}{45} = .6028$$

$$\eta_t = .9 \text{ (assumed)}$$

$$\frac{d\epsilon}{d\alpha} = .22, \quad \left(1 - \frac{d\epsilon}{d\alpha}\right) = .78$$

$$\tau = .6$$

$$c_{h_\alpha} = - .00527$$

$$c_{h_\delta} = - .01166$$

$$\left(1 - \tau \frac{c_{h_\alpha}}{c_{h_\delta}}\right) = .7288$$

Thus

$$\left.\frac{dC_M}{dC_L}\right|_{tail} = - \frac{.058}{.1078} \times .6028 \times .9 \times .78 \times .7288 = - \underline{\underline{.1659}}$$

The fuselage contribution is found from the integral

$$\frac{dM}{d\alpha} = \frac{q}{36.5} \int w_f^{\,2} \frac{d\beta}{d\alpha} \, dx$$

The integral was evaluated numerically:

$$\sum w_f^{\,2} \frac{d\beta}{d\alpha} \Delta x = 147.311$$

$$\left.\frac{dC_M}{dC_L}\right|_{fus} = \frac{\frac{dM}{d\alpha}}{qSc a_w}$$

$$= \frac{147.311/36.5}{120 \times 3.75 \times .1078}$$

$$= \underline{\underline{.0832}}$$

The stick free neutral point, or the c.g position at which the stability is zero, or neutral, is given by

$$N_o = A.C. - \sum \frac{dC_M}{dC_L}$$

$$= .25 + .1659 - .0832$$

$$= \underline{\underline{.3327 \text{ MAC}}}$$

The most aft c.g. from loading considerations is at .3184 MAC, therefore, this is satisfactory.

Control. - The elevator power is given by

$$C_{M_\delta} = - a_t \overline{V} \eta_t \tau$$

where

$$\overline{V} = {}^{29.51}/120 \times {}^{144}/45 = .787$$

$$C_{M_\delta} = .058 \times .787 \times .9 \times .6 = - .0246$$

For the GA(W)-1 airfoil with no flap (Reference 2)

$$\alpha_{OL} = - 4^\circ, \; c_{mac} = - .1, \; c_{l_{max}} = 1.77$$

With a .30 c Fowler flap deflected 40°, (Reference 3)

$$\alpha_{OL} = -20^\circ, \; c_{mac} = -.8, \; c_{l_{max}} = 3.80$$

thus,

$$\Delta\alpha_{OL} = -16^\circ, \; \Delta c_{mac} = - .70$$

the flapped area of the wing (b_f/b) is

$$S_{flap} = .8385 \; S_w$$

Therefore, for the three dimensional wing:

$$\alpha_{OL} = -4-16 \times .8385 = -17.4^{\circ}$$

$$C_{MAC} = -.1 - .7 \times .8385 = -.6870$$

and from the parametric landing program:

$$C_{L_{MAX}} = 2.861$$

The elevator angle for zero lift is given by

$$\delta_{e_o} = -\frac{C_{MAC}}{C_{M_\delta}} - \frac{(\alpha ol - i_w + i_t)}{\tau}$$

$i_w = i_t = 0$, therefore,

$$\delta_{e_o} = -\frac{.687}{.0246} + \frac{17.4}{.6}$$

$$= -27.9^{\circ} + 29^{\circ}$$

$$= \underline{1.1^{\circ}} \text{ (trailing edge down)}$$

The maximum stability level for stall is

$$\left.\frac{dC_M}{dC_L}\right|_{max} = \left(\delta_{e_o} - \delta_{e_{max}}\right) \frac{C_{M_\delta}}{C_{L_{MAX}}}$$

$$= (1.1 + 25)\,\frac{.0246}{2.861}$$

$$= \underline{.2244}$$

Thus the most forward c.g. allowable is

$$.3327 - .2244 = \underline{\underline{.1083 \text{ MAC}}}$$

Since the most forward c.g. position is .1148 MAC, this is satisfactory. Note that this is sufficient elevator power to stall out of ground effect, not on landing. Many small airplanes are not capable of stalling at forward c.g. with full flaps, even out of ground effect, and this is not an operational problem.

Lateral, directional, and dynamic stability analyses are not generally run until a later stage of the project (if at all).

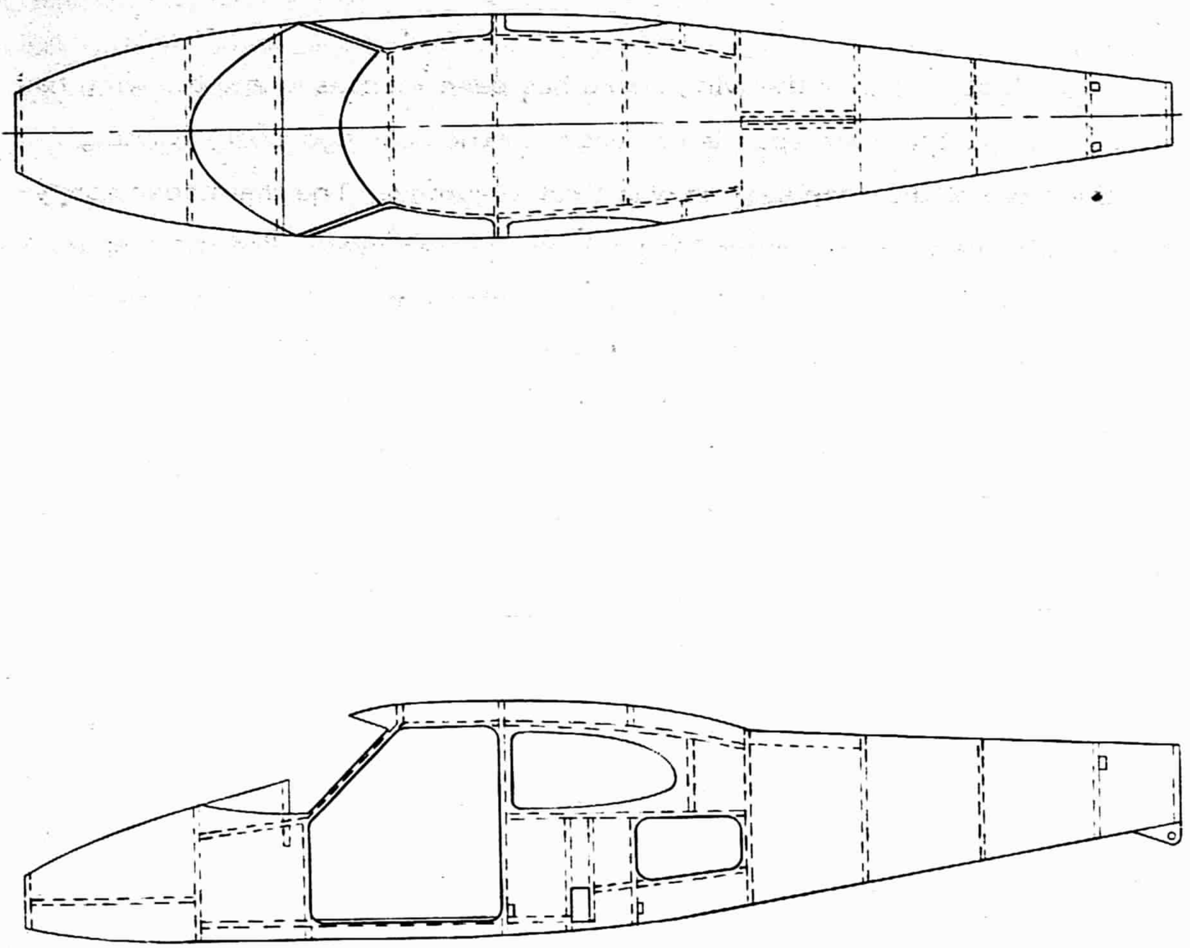
6.8 Structural Arrangement

Figure 35 shows a typical initial layout of the major structural elements. No loads have been calculated or members sized and conventional aluminum construction is assumed.

Fuselage. - Since the wing chord has been increased and the wing has been shifted aft, the main spar is no longer at the rear doorpost, running under the front of the rear seat as was first assumed. The main spar carrythru structure now passes under the back of the rear seat. The main spar carrythru structure carries the wing bending moments and the main gear attachments. It can be constructed either of extruded caps and vertical stiffeners with sheetmetal webs and forged landing gear sockets, or, if a large numerically controlled milling machine is available, it can be a single large forging. Since the carrythru is located in the middle of the rear window, the vertical shear loads must be carried by the skins forward to the doorpost frame. Wing torsion loads are carried by fore and aft fittings.

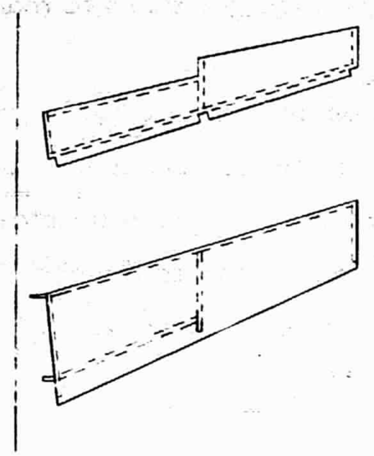
The main engine mount is carried by the aft cabin bulkhead. The stabilizing link is attached to a fore and aft beam on the top centerline. This mounting structure should be stressed for about 25 g's in a forward and downward direction for crash protection of the occupants. This should not be too difficult since the engine is very light.

The last two tailcone bulkheads carry the tail surface attachments. The aft bulkhead carries the bending loads and the next bulkhead carries the torsion. The aft bulkhead also carries the tailskid/tiedown fitting loads. The tailcone skins are simple flat wrapped sheetmetal. An analysis would be required to determine the need for stringers. These could be bent up on the edges of the skins if the panel curvature is not enough to stabilize the skins.

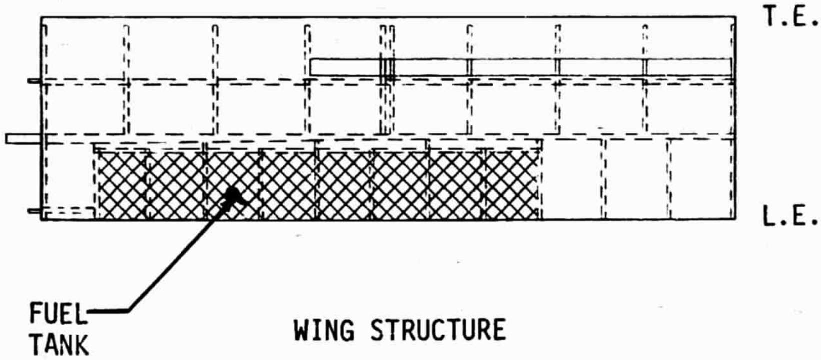


FUSELAGE STRUCTURE

FIGURE 35 - Structural Arrangement



TAIL STRUCTURE



WING STRUCTURE

FIGURE 35 (Concluded)

The nose, from the forward cabin bulkhead forward, could be a simple plastic fairing; however, a metal structure is preferred to provide material for crash energy absorption, in spite of the higher cost. An additional baggage compartment could be located there to make use of this space.

<u>Wing.</u> - Bending loads are carried by the main spar, which is located at 40% chord. The inboard portion of the spar is an extrusion with tapered caps while the outboard portion is a simple sheetmetal channel. The aft spar closes out the flap bay and supports the control systems. The forward spar is provided to achieve another shear fitting and only extends outboard one bay, to the inboard end of the fuel tank.

The fuel tank is a sealed portion of the leading edge. A closeout spar is provided ahead of the main spar which eliminates the need to seal to the tapered extrusion. The fuel tank can thus be assembled, sealed, and tested as a unit.

The long travel of Fowler flaps require correspondingly long support tracks. If these tracks are external to the contour, they require large fairings and are objectionable from both drag and appearance standpoints; therefore, they are contained as completely as possible within the wing contour. This requires splitting each wing flap into two segments, with the rollers on the ends riding on three tracks. Since the wing is not tapered, the four flap segments could be identical and interchangeable. Each segment must be actuated at each end to avoid binding in the tracks, therefore, three bell cranks are required in each wing. A major design problem associated with this type of flap is the support of the slot lip to maintain the proper gap. Possibly, the simplest solution is a sandwich type construction using a high density, poured in place, foam core.

The spoilers are of triangular cross section for maximum torsional and bending stiffness. These can be folded from single pieces of sheetmetal with hinge provisions added at each rib.

Tail surfaces. - The stabilizer aft spar is a sheetmetal channel, with doublers in the inboard end as required to carry the bending loads; the root fitting is a forging. The forward spar transmits the torque and is only half span. The skin is one piece.

The elevator (or ruddervator) also features a one piece skin, either wrapped around the leading edge and riveted at the trailing edge, or folded at the trailing edge and riveted to the spar at the leading edge. The control horn at the inboard end is a steel weldment. A trim tab is provided on each elevator and the mass balance weights are incorporated in the tips. The stabilizers, elevators, trim tabs, and tips may be made identical and interchangeable.

All extremities of the airplane - the nose cap, tailcone fairing, wing tips, and tail tips, are made of plastic, as well as other fairings such as wheel fairings and wing fillets. Whether these are fiberglass layups or thermoformed ABS would depend on the capabilities and economics of the individual factory.

6.9 Costs

Aircraft costs were examined for development and certification, production, initial pricing, and annual operations. Cost analysis methodology for estimating development and production costs employed the contemporary AMPR (Airframe Manufacturer's Production Responsibility) method in determining a parameter to which direct labor manhours could be associated through engineering and manufacturing historical experience.

The AMPR weight method for associating manufacturing manhours has been found to be a convenient and reasonably accurate method for predicting costs in the preliminary design process. During the program definition phase, just prior to launch of a new airplane development, most manufacturers will develop detailed task and manhour schedules to which specific skills and organizations are assigned. With this information a more accurate and time-phased cost analysis can be made.

For the present analysis, the detail methodology used is presented in Appendix C. Only the selected contemporary design aircraft described in previous sections was evaluated for cost. Development, production, and operations cost were based on constant early CY '77 dollars. Production costs were based on cumulative average manhours of a mature manufacturing operation wherein minimum manhours per unit weight of AMPR weight were achieved. Operations costs were based on conventional estimating techniques used by the General Aviation Industry.

A summary of Development and Certification Costs is presented in Table 4 showing the major items of costs within Engineering, Tooling, Quality Assurance, and Manufacturing. In the latter case, manufacturing involves construction of only prototype and structural test articles.

TABLE 4

DEVELOPMENT AND CERTIFICATION COST

(Early CY77 Dollars)

Engineering		
	Engineering Burdened Labor	$ 1,308,288
	Special Materials & Purchased Services	250,000
	Flight Testing (500 Hours)	25,000
	Total Engineering Cost	$ 1,583,288
Tooling		
	Direct Labor	$ 392,889
	Overhead	404,676
	Materials	96,612
	Total Tooling Cost	$ 894,177
Manufacturing (1 Prototypes; 2 Test Articles)		
	Direct Labor	$ 63,780
	Overhead	86,103
	Materials	22,172
	Total Manufacturing Cost	$ 172,055
Quality Assurance		
	Direct Labor	$ 30,093
	Overhead	40,626
	Total Quality Assurance Cost	$ 70,719
	Total Development & Certification Cost	$ 2,720,239

Production Unit Cost and Estimated Pricing are presented in Table 5. The manner of presentation of these costs in Table 5 are a result of the estimation method. Out-the-Factory-Door (OFD) costs include Production Unit Costs, Period Costs, and Warranty Reserve, which are estimated as a percentage of Production Unit Cost.

Aircraft estimated price is shown at the bottom of Table 5 both with and without amortization of development costs. The estimated price does not include add-on equipment. Write-off of development costs varies widely among manufacturers depending on their financial health, the market, and management decisions. For high performance business jet airplanes development costs may be written off over only 50-100 units, while in small aircraft it may be over several thousand units. In some instances initial pricing of medium twin-engine aircraft have been established to provide development write-off over only 15-25 units. For this study 3000 units was assumed.

Estimated Cost of Operations are presented in Table 6 showing breakdown of Variable and Fixed Costs. Costs shown are for an annual utilization based on engine manufacturer's estimated costs for the engine design employed. Estimated TBO versus utilization is listed as follows.

Annual Utilization (Hrs/Yr)	Estimated TBO (Hrs)
75	1125
100	1475
150	2100
200	2575
250	2875
300	3000

TABLE 5

PRODUCTION UNIT COST
(Early CY77 Dollars)

Manufacturing Materials	
Engine	$12,500
Basic Avionics and Other	5,418
Manufacturing Direct Labor	2,846
Manufacturing Overhead	4,554
Quality Assurance Direct Labor	302
Quality Assurance Overhead	483
Total Production Unit Cost	$26,103

ESTIMATED PRICE
(Mid-FY77 Dollars)

Production Unit Cost	$26,103
Period Costs	2,610
Warranty Reserve	522
Gross Margin	15,742
Total Price (W/O Amortization of Development Costs)	$44,977
Development Amortization	907
Total Price (Including Development Amortization)	$45,884

TABLE 6

COST OF OPERATION

(Early CY77 Dollars Without Amortization)

At 200 hrs/yr.

Purchase Price	$ 44,977
Cruise Speed	162 MPH
Miles Per Year	32,400 S.M.
Variable Cost/Hour	
Fuel & Oil [(18.05 X .67) + .60]	$ 12.69
Airframe & Avionics Maintenance Reserve	8.40
Engine Overhaul and HSI	2.00
Parking/Landing Fees & Spare Parts Inventory	1.17
Total Variable Cost/Hr.	$ 24.26
Total Variable Cost/Yr.	$4852.00
Fixed Cost/Yr.	
Depreciation	$ 4,498
Crew	0
Insurance	
Hull	360
Liability/Medical	325
Hangar/Tie Down	450
Navigation Materials	100
Airways Tax	25
Total Fixed Cost/Yr.	$ 5,758
Total Operating Cost/Yr.	$ 10,610
Total Operating Cost/Hr.	$ 53.05
Total Variable Cost/S.M.	$ .150
Total Operating Cost/S.M.	$ .328

For annual utilization under 75 hours per year, overhaul would occur every seven years. Overhaul cost is based on 40 percent of original engine cost.

Depreciation is based on an 8-year period with value diminishing on a straight line basis to 20 percent. Other cost factors of variable and fixed costs are determined by conventional estimating methods.

7.0 GASP AIRPLANE DESIGN

7.1 Iterations and Evolution of the Design

The design process using GASP is started from a 3-view drawing and a baseline definition of the aircraft. Depending on the level of technology employed, those items which can be defined as known or required are identified. For example, if a known landing gear, engine or wing planform is to be used, these items are input as fixed and will not be varied or scaled. If performance items are identified as requirements, these, too, are fixed and the final design will be sized to meet these constraints. The 3-view is used primarily to define the geometry and provide a valid starting point.

The design process utilizing the GASP program was initiated by duplicating the geometry and component weights of the PD1502 design of section 6.6. This was done to calibrate the coefficients in the weight trend equations to represent this class of aircraft.

The PD1502 iarcraft was used as the starting point for this design exercise. The aircraft was sized with the following constraints, some of which were carried over from the previous Garrett study:

Airframe requirements:

1) Cabin size fixed (4 seats)
2) Fixed equipment weight 97.1 Kg (214 lbs.)
3) Design payload (2 passengers + 1 crew) 272 Kg (600 lbs.)

Mission requirements:

1) Cruise @ 3048 m (1000 ft.) @ 241 KPH (150 MPH)
2) Range 1482 Km (800 NM) with 45 min. reserve
3) Takeoff and landing distance = 610 m (2000 ft.)

Engine requirements:

1) Garrett small turbofan engine
2) Engine cost 25$/lb thrust
3) 3000 Hr. TBO

All other aspects of the design could be varied as desired. The baseline aircraft defined by the GASP program with these constraints and the PD1502 geometry is given in the computer output in Table 8 (Run 1).

7.2 Sensitivity Analysis

After the various weight coefficients had been selected to represent the baseline aircraft, those pertinent parameters which may be scaled were varied throughout their practical range to determine the effect on the design.

For the sensitivity study, the baseline aircraft was varied for takeoff and landing distance, aspect ratio, wing loading, thickness ratio, wing sweep, incremental weight and drag, and mission range. Table 7 shows the ranges of data investigated for these variable parameters. It was determined during the course of the study that the flap methodology was only good up to an aspect ratio of 12. Several runs, including the final run, were made at aspect ratios beyond this value; in these cases, the results reflect an aspect ratio of 12.

TABLE 7.-VARIABLE PARAMETERS

Parameter	Range
Aspect Ratio	6 - 16
Wing Loading	97.6 -146.5 KG/M^2 (20 -30 Lb/Ft^2
Thickness Ratio	8 - 21%
Sweep	$-5^\circ - 10^\circ$
αOL	$-3.9^\circ - 0^\circ$
Δ Weight	$\pm$ 45.4 KG (100 Lb)
Δ Drag	$\pm$ 30 Counts

By comparing various criteria such as gross weight, wing area, mission fuel, static thrust, cost, etc., as affected by changes in the variable parameters, the sensitivity of the design can be evaluated. Likewise minimum or maximum points can be determined if they exist in the tested range of the parameter. Table 8 lists the parameters tested and the effects compared to the baseline design.

TABLE 8.-GASP DESIGN ITERATIONS

RUN NO.	AR	WING LOADING	THICKNESS RATIO	SWEEP	T.O./LDG DISTANCE	ΔWT	ΔCD	α ZERO LIFT	RANGE	SLS THRUST	WING AREA	GROSS WEIGHT	MISSION FUEL	COST
		Lbs/Ft²	%	Deg	Ft	Lbs	Counts	Deg	NM	Lbs	Ft²	Lbs	Lbs	$
1 (Baseline)	12	25	17	0	2000	0	0	-3.9	800	441	84.4	2110	463	27918
2	12	25	17	0	1750	0	0	-3.9	800	540	88.9	2222	508	33075
3	12	25	17	0	1500	0	0	-3.9	800	700	95.5	2388	569	41452
4	10	25	17	0	2000	0	0	-3.9	800	456	85.8	2144	490	28620
5	8	25	17	0	2000	0	0	-3.9	800	501	88.9	2222	537	30935
6	6	25	17	0	2000	0	0	-3.9	800	565	93.4	2334	603	34261
7	14	25	17	0	2000	0	0	-3.9	800	443	84.3	2107	454	28046
8	16	25	17	0	2000	0	0	-3.9	800	448	84.6	2116	451	28442
10	12	20	17	0	2000	0	0	3.9	800	415	111	2222	402	27621
12	12	25	17	0	2000	0	0	-2.0	800	493	86.9	2174	489	30662
13	12	25	17	0	2000	0	0	0	800	558	89.5	2238	513	33990
14	12	30	17	0	2000	0	0	-3.9	800	562	72.4	2171	507	33554
(New Baseline)														
21	12	25	17	0	2000	0	0	-3.9	800	458	85.1	2126	468	28773
22	12	25	21	0	2000	0	0	-3.9	800	446	84.3	2108	470	28034
23	12	25	13	0	2000	0	0	-3.9	800	531	89.5	2237	505	32848
24	12	25	08	0	2000	0	0	-3.9	800	720	100.4	2510	589	43442
25	12	25	17	-5	2000	0	0	-3.9	800	461	85.1	2129	469	28934
26	12	25	17	+5	2000	0	0	-3.9	800	461	85.2	2130	470	28953
27	12	25	17	10	2000	0	0	-3.9	800	476	86.5	2162	486	29779
28	12	25	17	0	2000	-20	0	-3.9	800	449	83.3	2082	460	28016
29	12	25	17	0	2000	+20	0	-3.9	800	467	86.3	2169	475	29518
30	12	25	17	0	2000	50	0	-3.9	800	479	89.8	2246	494	30675
31	12	25	17	0	2000	100	0	-3.9	800	500	93.7	2342	504	32460
32	12	25	17	0	2000	-50	0	-3.9	800	437	80.8	2021	452	26951
33	12	25	17	0	2000	-100	0	-3.9	800	414	76.4	1910	433	25047
34	12	25	17	0	2000	0	+10	-3.9	800	464	86.1	2153	483	29146
35	12	25	17	0	2000	0	20	-3.9	800	472	87.7	2194	508	29659
36	12	25	17	0	2000	0	30	-3.9	800	487	88.9	2222	521	30503
37	12	25	17	0	2000	0	-10	-3.9	800	452	84.2	2105	457	28440
38	12	25	17	0	2000	0	-20	-3.9	800	447	83.2	2081	443	28092
39	12	25	17	0	2000	0	-30	-3.9	800	443	82.6	2065	435	27848
40	12	25	17	0	2000	0	0	-3.9	400	377	69.1	1729	220	23717
41	12	25	17	0	2000	0	0	-3.9	479	391	72.0	1800	265	24605
42	12	25	17	0	2000	0	0	-3.9	600	414	76.5	1912	336	26014
43	12	25	17	0	2000	0	0	-3.9	700	436	80.7	2018	401	27386
44	12	25	17	0	2000	0	0	-3.9	900	482	90.3	2258	550	30340
45	12	25	17	0	2000	0	0	-3.9	1000	510	95.5	2388	629	32138
46 Final	13	22.5	21	0	2000	0	0	-3.9	800	411	94.2	2120	456	26708

It should be noted that the baseline design changed slightly at Run No. 21. The new baseline parameters are shown in Table 8. At this point a keypunch error had occurred in defining the parameter VKTIN in the flap definition. The original baseline used VKTIN = 80 knots and runs 1 through 20 were made using this value. In Run No. 22, VKTIN = 60 knots was used and this value was carried through Run No. 45. When the error was discovered, a new baseline run was made with this value of VKTIN and the baseline 2 aircraft was defined. The actual differences between the two baselines are small and it was not deemed necessary to rerun the sensitivity studies. The data from Runs 22-45 is compared to the baseline 2 aircraft.

Three design criteria were chosen to evaluate the effects of the variables on the design. These are gross weight, engine static thrust, and retail cost. Plots of the effects of the variable parameters on these criteria are shown in Figures 36 thru 38. It can be seen that the trends of most of the variable parameters do not show minimum or maximum characteristics; however, aspect ratio, wing loading, thickness ratio and sweep show either a minimum value or at least a flat trend. Wing sweep effects are nearly flat for small sweep angles. Aspect ratio and thickness ratio show a flattening trend at higher values without a well-defined minimum in the range tested. Wing loading, however, not only shows a definite minimum, but the optimum wing loading is different for each of the criteria of gross weight, static thrust and cost.

A second design point was optimized which matches closely the PD1502A design selected in Section 6.6. The only differences from the requirements of the first design are in the mission performance criteria. These new performance criteria were:

Range = 885 KM (479 NM) (550 SM) with 45 min. reserve

Takeoff and landing distance - 457 M (1500 Ft.)

Based on these new constraints, a second sensitivity study was made. This study was more limited in scope and involved only the variable parameters of wing loading, aspect ratio and thickness ratio. Table 9 lists the results of the sensitivity study. The plots of the effects on the criteria of gross weight, static thrust and cost are shown in Figures 39 thru 41. Although wing loading tends to reach minimum values, these curves are somewhat flatter than the corresponding ones of the previous sensitivity study.

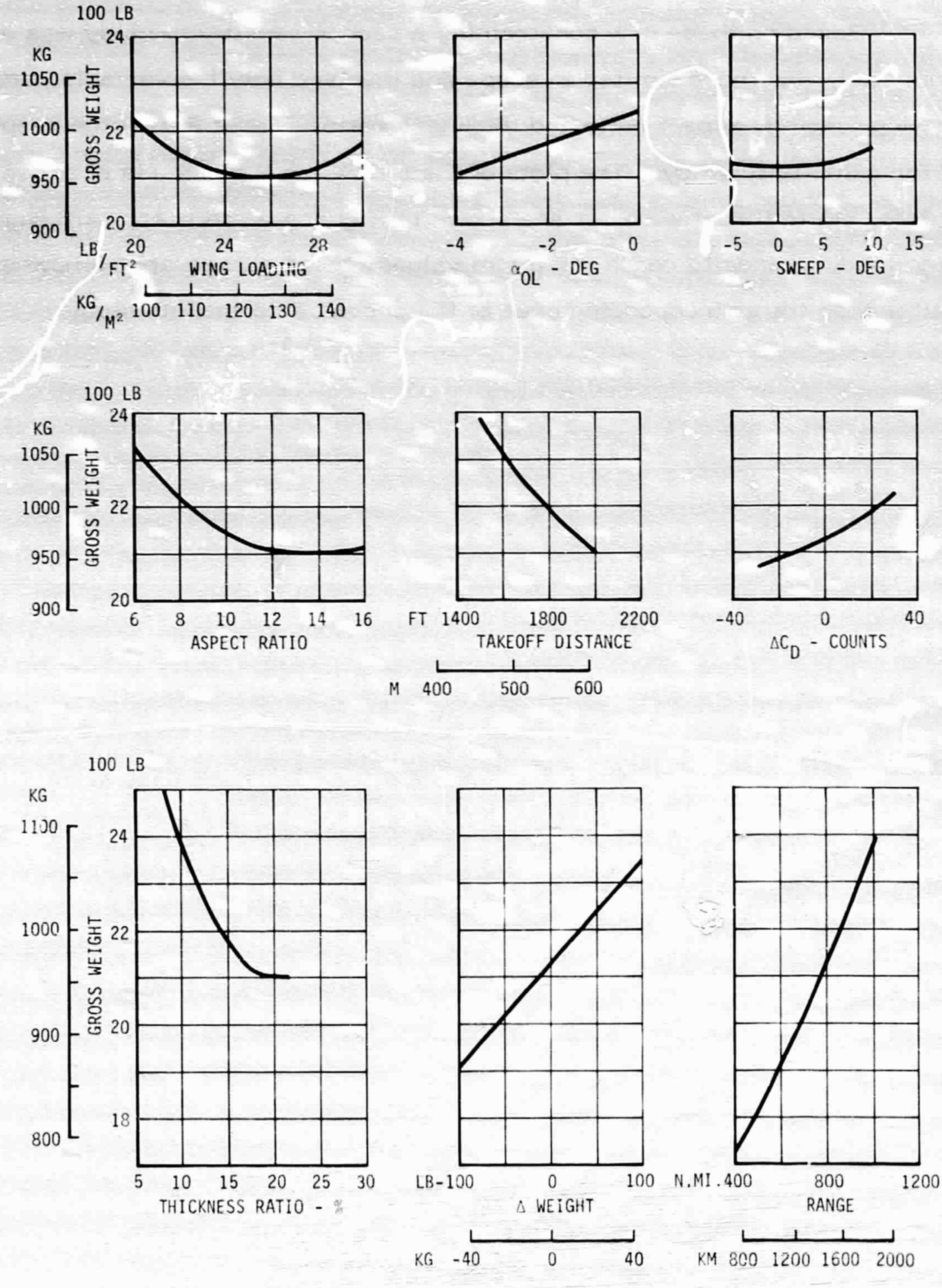


FIGURE 36 - EFFECT OF VARIABLE PARAMETERS ON GROSS WEIGHT

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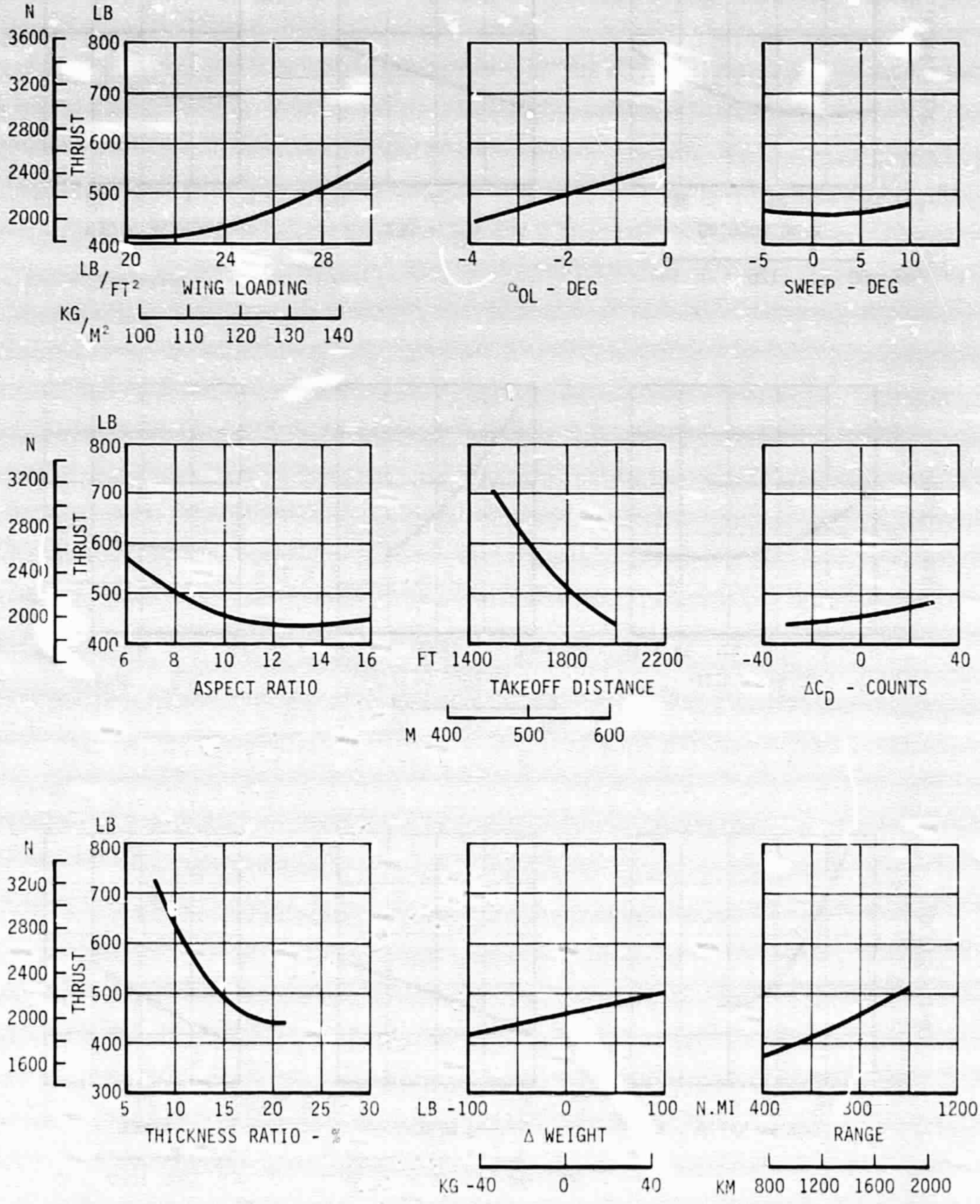

FIGURE 37 - EFFECT OF VARIABLE PARAMETERS ON SEA LEVEL STATIC THRUST

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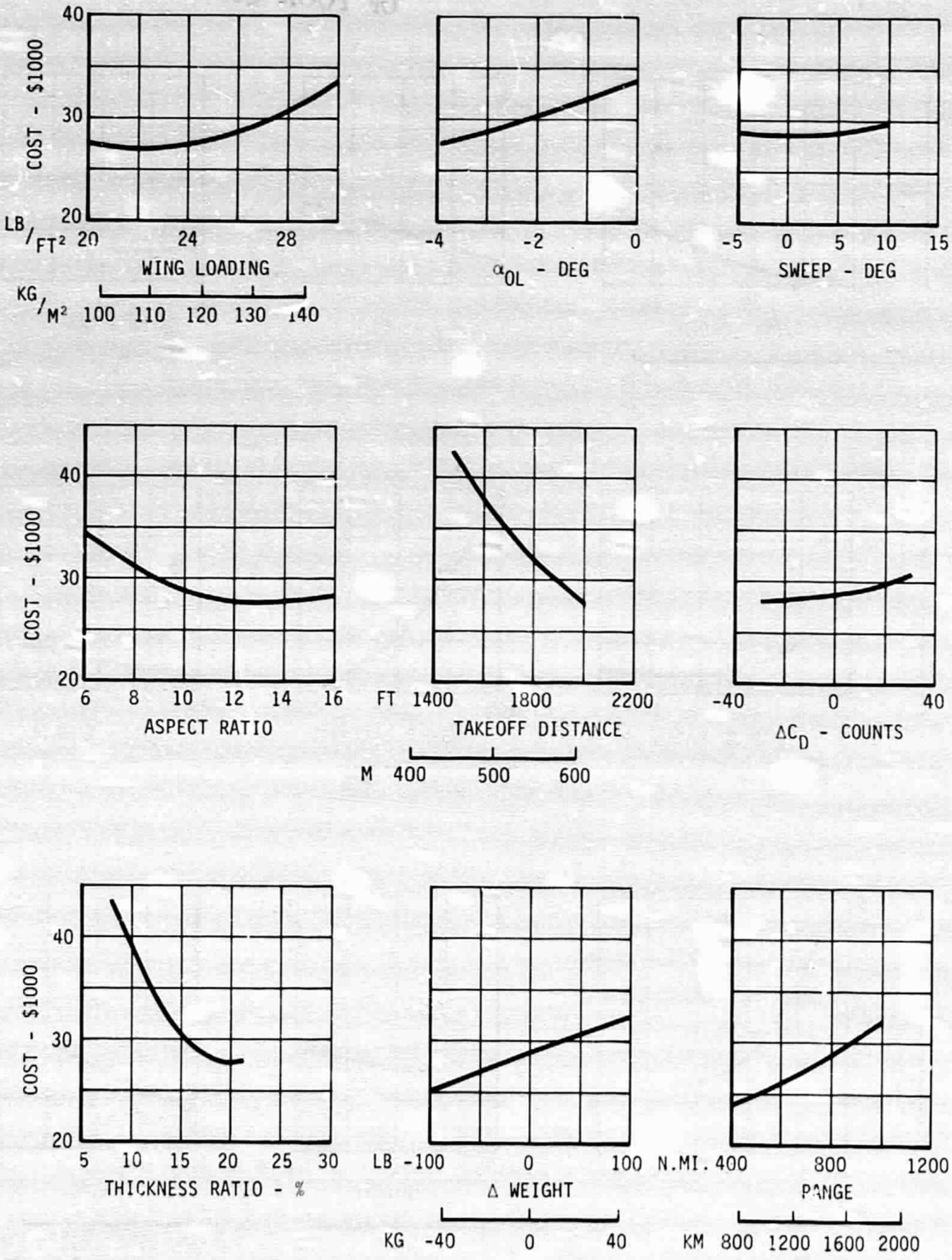

FIGURE 38 - EFFECT OF VARIABLE PARAMETERS ON COST

TABLE 9. -GASP ITERATIONS

RUN NO.	ASPECT RATIO	WING LOADING	THICKNESS RATIO	T.O./LDG DISTANCE	RANGE	SLS THRUST	WING AREA	GROSS WT	MISSION FUEL	COST
		Lbs/Ft^2	%	Ft	S.M. (479NM)	Lbs	Ft^2	Lbs	Lbs	$
50 (Baseline)	12	25	17	1500	550	581	78.0	1949	309	34245
51	12	25	21	1500	550	557	76.6	1916	302	32877
52	12	25	13	1500	550	685	82.4	2060	337	39838
53	12	25	8	1500	550	985	94.7	2369	414	56040
54	12	30	17	1500	550	747	67.5	2026	345	42213
56	12	22.5	17	1500	550	516	86.4	1944	299	31313
57	12	20	17	1500	550	456	97.4	1948	293	28674
60	10	25	17	1500	550	599	78.8	1971	324	35124
61	8	25	17	1500	550	634	80.3	2007	345	36789
62 Final	12	20	21	1500	550	437	95.8	1916	287	27573

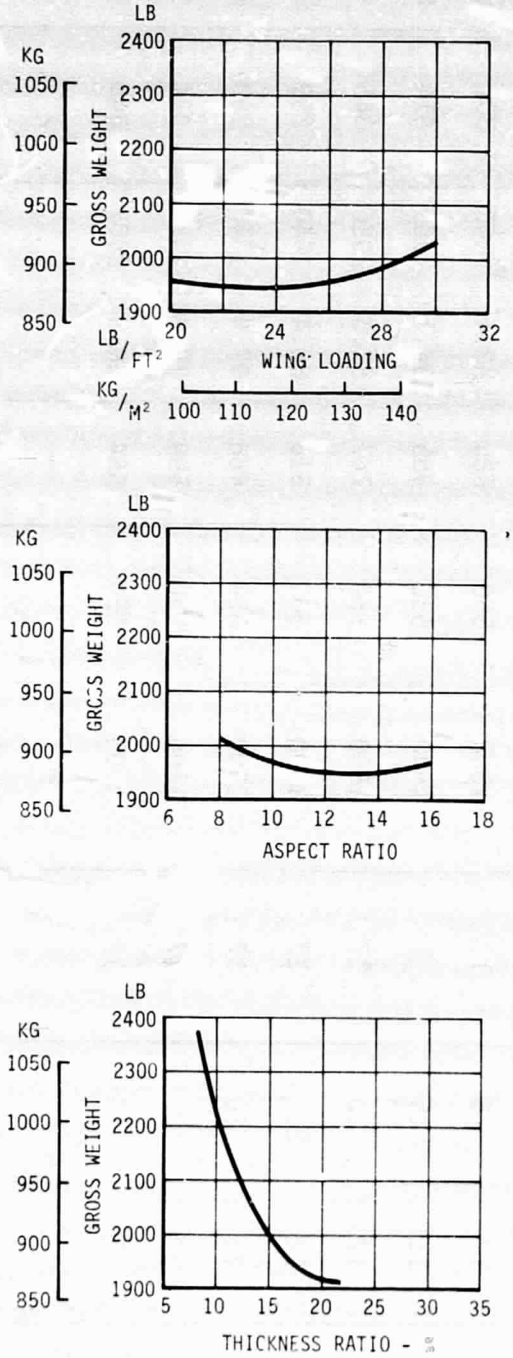

FIGURE 39 - EFFECT OF VARIABLE PARAMETERS ON GROSS WEIGHT

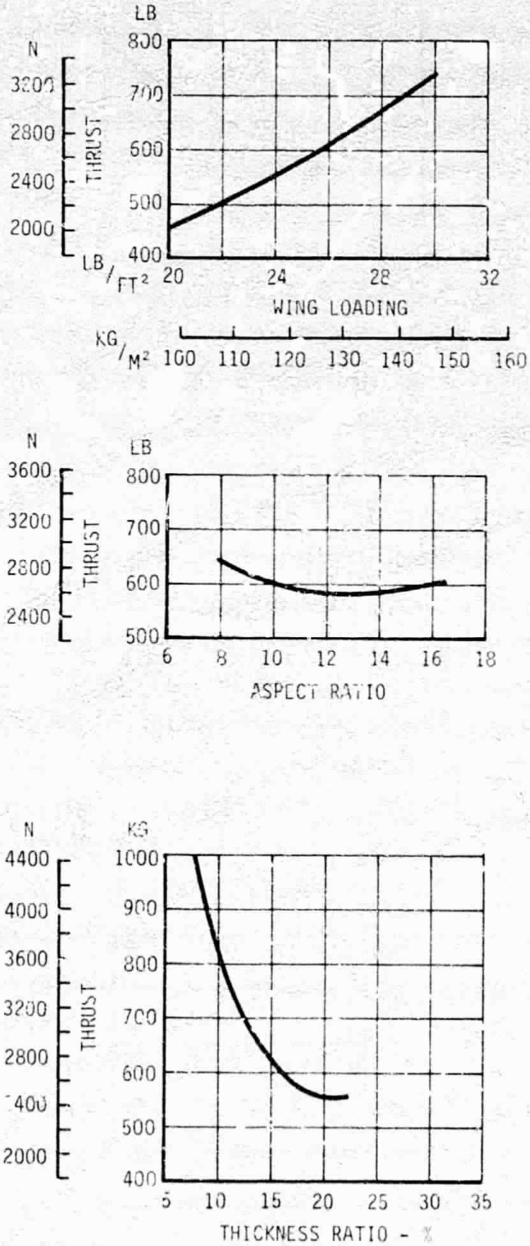

FIGURE 40 - EFFECT OF VARIABLE PARAMETERS ON SEA LEVEL STATIC THRUST

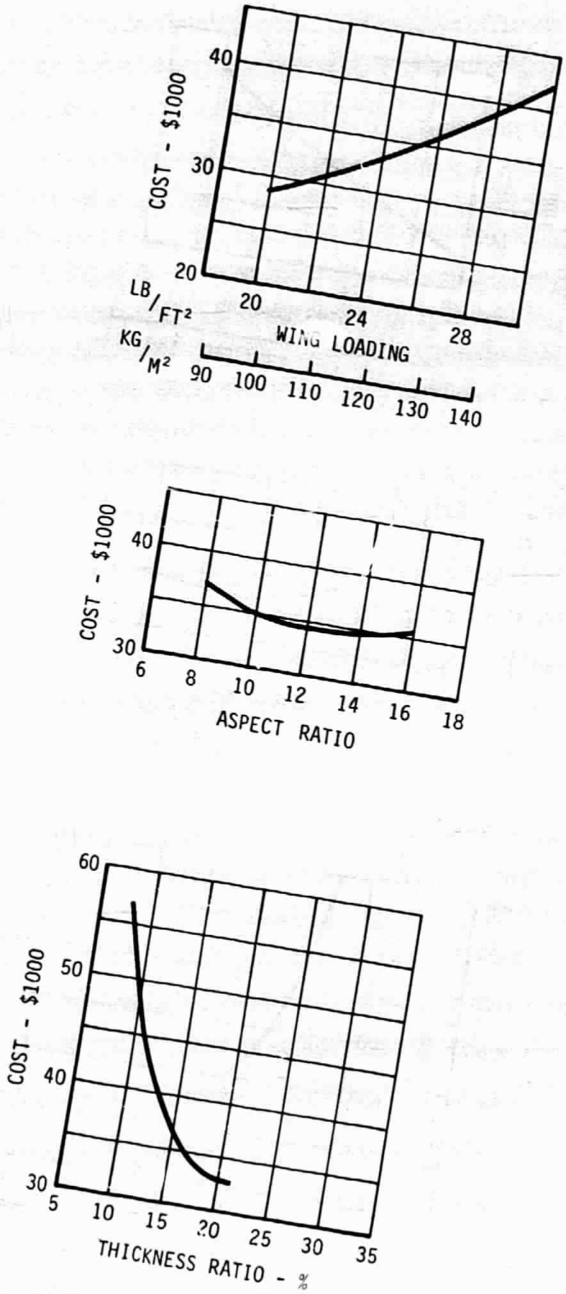

FIGURE 41 - EFFECT OF VARIABLE PARAMETERS ON COST

C-2

7.3 Final Design Configuration

The final design point selected for each of these two optimizations was based on retail cost. The following values were selected to represent the final design and the data input for the final design computer run:

Takeoff Distance	610M (2000 Ft)	457M (1500 Ft)
Range	1481 KM (800 NM)	887 KM (479 NM)
Wing Loading	110 KG/M^2 (22.5 PSF)	97.6 KG/M^2 (20 PSF)
t/c	21%	21%
AR	13	12

The final designs based on these criteria are described by the GASP output shown in Tables 10 and 11. (Run No.'s 46 and 62, respectively.)

TABLE 10.-GASP FINAL DESIGN (RUN 46)

Takeoff Gross Weight	962 KG (2120 Lb)
Wing Area	8.75 M^2 (94.2 Ft^2)
Sea Level Static Thrust	1828 N (411 Lb)
Mission Fuel	207 KG (456 Lb)
Retail Cost	$26708.
2 Passengers + 1 Crew	
Wing Loading	109.9 KG/M^2 (22.5 Lb/Ft^2)
Aspect Ratio	13
Thickness Ratio	21%
Sweep	0^o
OL	-3.9^o
Takeoff and Landing Distance	610 M (2000 Ft)
Cruise Altitude	3048 M (10000 Ft)
Cruise Speed	241 KPH (150 MPH)
Range	1481 KM (800 NMI)

TABLE 11.-GASP FINAL DESIGN (RUN 62)

Takeoff Gross Weight	869 KG (1916 Lb)
Wing Area	8.90 M^2 (95.8 Ft^2)
Sea Level Static Thrust	1944 N (437 Lb)
Mission Fuel	130 KG (287 Lb)
Retail Cost	$27573.
2 Passengers + 1 Crew	
Wing Loading	97.6 KG/M^2 (20 Lb/Ft^2)
Aspect Ratio	12
Thickness Ratio	21%
Sweep	0^o
OL	-3.9^o
Takeoff and Landing Distance	457 M (1500 Ft)
Cruise Altitude	3048 M (10000 Ft)
Cruise Speed	241 KPH (150 MPH)
Range	887 KM (479 NMI)

8.0 COMPARISON BETWEEN THE CONVENTIONAL AND GASP DESIGN PROCEDURES

8.1 Introduction

The conventional design sensitivity studies calculated a series of gross weights for aircraft with various wing spans and areas as shown in Figure 6. Using the weight coefficients determined for the GASP baseline aircraft along with gross weight, wing span and area from Figure 6, performance calculations are compared between the conventional analysis and the GASP program. It should be noted that the aerodynamics and certain performance constraints used in the GASP runs were not the default aerodynamics of GASP, but rather were the default values used in the previous Garrett study. Table 12 shows the specific comparison points selected and the values of input data to the GASP program. To evaluate the performance methodology, the GASP output is used directly and plotted to the same scale and format as shown in the corresponding plots of the conventional analysis. All of these comparisons are for a 400 pound static thrust engine. Table 13 lists the data from GASP used in the following plots.

Flaps up stall speed, Figure 42, shows the same trends as the conventional analysis, Figure 7, and gives substantially the same speeds as a function of wing area and span.

The takeoff ground roll shown in Figures 43 and 16 has the same trend, but the distances predicted by GASP are longer than those of the conventional analysis. Total distance to 15 m (50 ft) does not exhibit the same curve shape for the two methods, nor do the distances agree. The conventional method, Figure 18, shows a definite minimum as a function of wing area. In Figure 44, the GASP method does not exhibit this characteristic but continues to reduce with increasing wing area. The distances calculated by the GASP

program are from 20% to 80% longer than the conventional analysis. In the takeoff and landing calculations, better agreement would have been obtained by inputting the same speed margins, time delays, and load factor requirements into GASP as used in the conventional analysis. Input parameters are available for doing this, however, the Garrett values were retained.

The GASP cruise, Figures 45-48, do not compare directly to the conventional analysis cruise of Figures 11-14. At the time the study was performed, GASP did not calculate V_{max} directly could it perform a cruise at a fixed power setting such as maximum cruise thrust or maximum thrust. GASP cruises were limited to fixed cruise Mach number for a given altitude. Maximum range cruise data of Figures 46-49 is calculated by computing cruise at several speeds, and crossplotting to determine max range speed. Figures 49-52 show the specific range plots for the various combinations of wing span and area.

Figure 53 shows the range capability of these various design points as a function of wing span and area for the design mission of 3048 m (10,000 ft) cruise at 241 kph (150 mph). The mission fuel for the GASP analysis is dependent on the gross weight, whereas the conventional analysis uses a fixed fuel available.

Landing distance of 15 m (50 ft) is shown in Figure 54. In contrast to the takeoff distance, the GASP method produces values significantly closer to those of the conventional method, with the variation being from 10% to 25%. This, again, is due primarily to the GASP inputs retained from the Garrett study.

TABLE 12.-METHODOLOGY EVALUATION POINTS

(400 Lb. Thrust Engine)

Wing Span Ft	Wing Area Ft^2	Aspect Ratio	Gross Weight Lbs	Wing Loading Lbs/Ft^2
40	400	4	2078	5.195
	267	6	2080	7.79
	200	8	2082	10.41
	133	12	2084	15.67
35	306	4	2038	6.66
	204	6	2040	10.0
	153	8	2042	13.35
	102	12	2045	20.05
30	225	4	2000	8.89
	150	6	2002	13.35
	112	8	2004	17.89
	75	12	2006	26.75
25	156	4	1962	12.58
	104	6	1964	18.88
	78	8	1966	25.20
	52	12	1969	37.86

TABLE 13.-GASP COMPARISON

RUN NO.	SPAN	A/R	AREA	TOGW	STALL SPEED	R/C SL	MAX RANGE SPEED CRUISE SL	CRUISE 5000	CRUISE 10000	CRUISE 15000	T.O. GRND ROLL	T.O. TO 50 FT	LANDING 50 FT	RANGE 10000 150MPH	COST
	Ft		Ft^2	Lbs	MPH	FPM	MPH	MPH	MPH	MPH	Ft	Ft	Ft	N.M.	$
70	40	4	400	2078	38.9	676	99	109	113	116	633	1652	541	24	46609
71		6	267	2080	47.7	770	107	114	120	127	908	1 18	651	217	43113
72		8	200	2082	54.2	831	111	119	128	135	1165	2226	734	371	41386
73		12	133	2084	65.8	904	119	127	137	145	1665	2926	854	567	39699
75	35	4	306	2038	44.8	721	108	117	123	127	780	1940	574	132	43192
76		6	204	2040	54.2	814	116	124	133	139	1147	2270	695	343	40561
77		8	153	2042	61.7	871	120	129	141	146	1457	2653	786	475	39378
78		12	102	2045	74.8	938	126	136	148	155	2063	3532	928	642	38132
80	30	4	225	2000	51.6	759	120	132	137	141	1023	2381	625	273	40219
81		6	150	2002	62.8	848	126	139	148	151	1485	2806	762	454	38449
82		8	112	2004	71.8	899	134	145	154	157	1887	3320	870	563	37622
83		12	75	2006	86.7	818	142	150	163	167	2782	4486	1045	707	36544
85	25	4	156	1962	51.7	781	137	147	151	153	1417	3134	711	398	37726
86		6	104	1964	75.2	860	140	154	161	163	2032	3722	875	527	36734
87		8	78	1966	85.9	909	149	160	166	172	2585	4433	1008	612	36084

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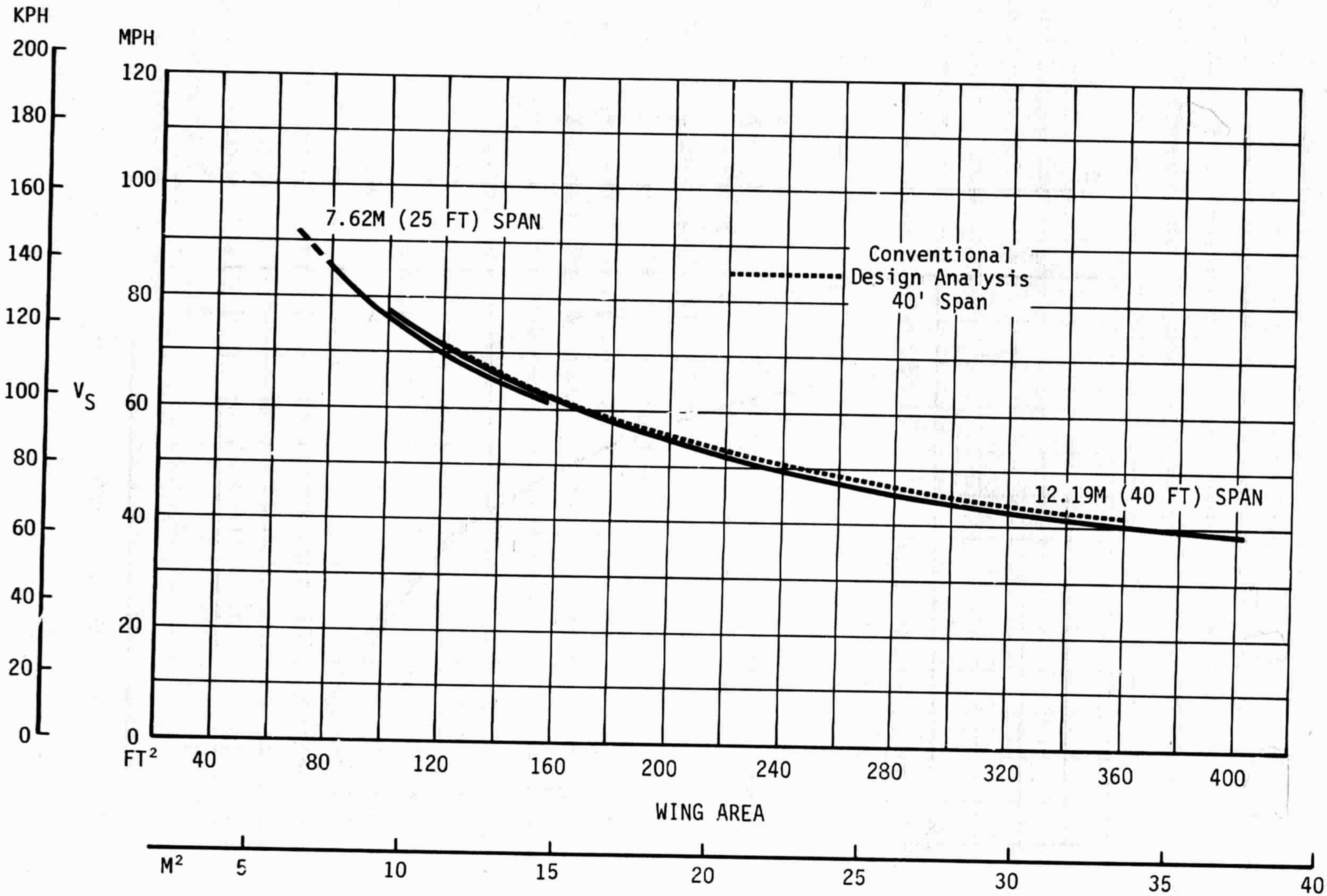

FIGURE 42 - FLAPS UP STALL SPEED

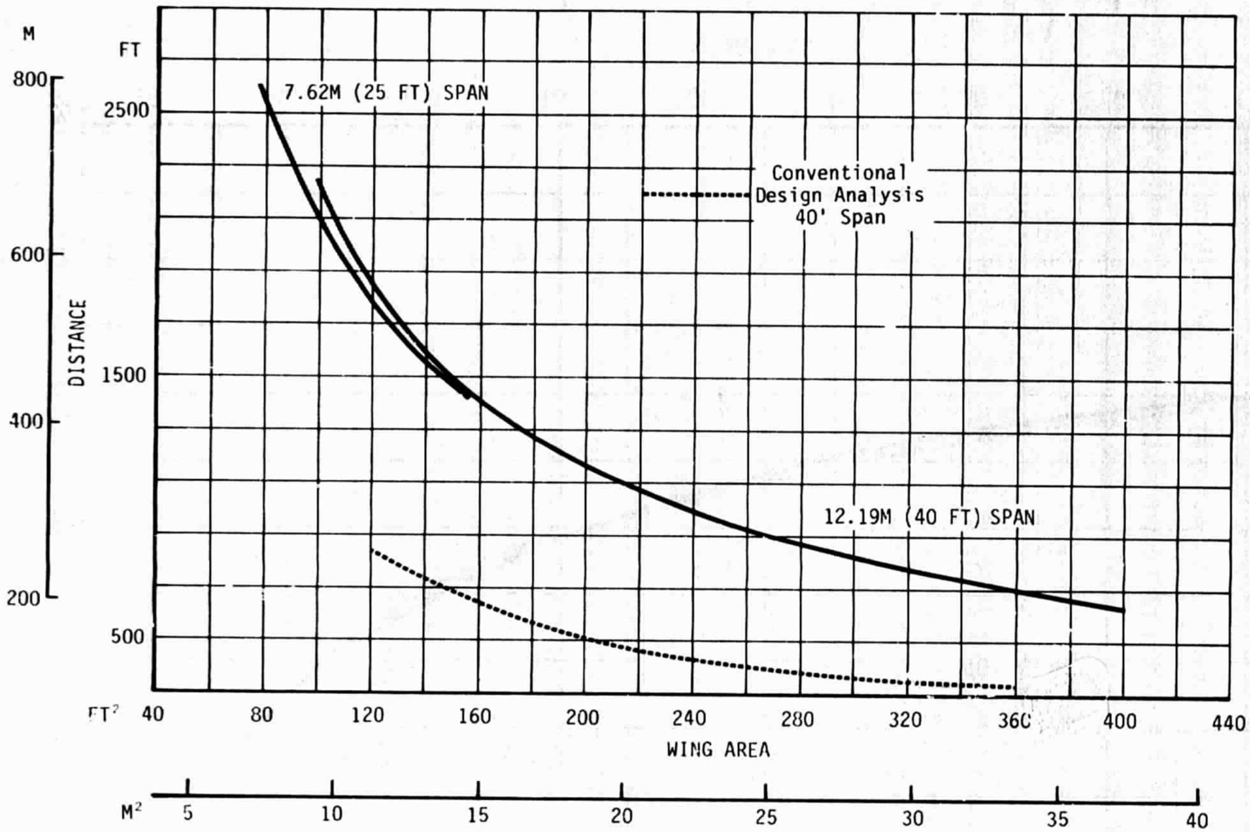

FIGURE 43 - TAKEOFF GROUND ROLL

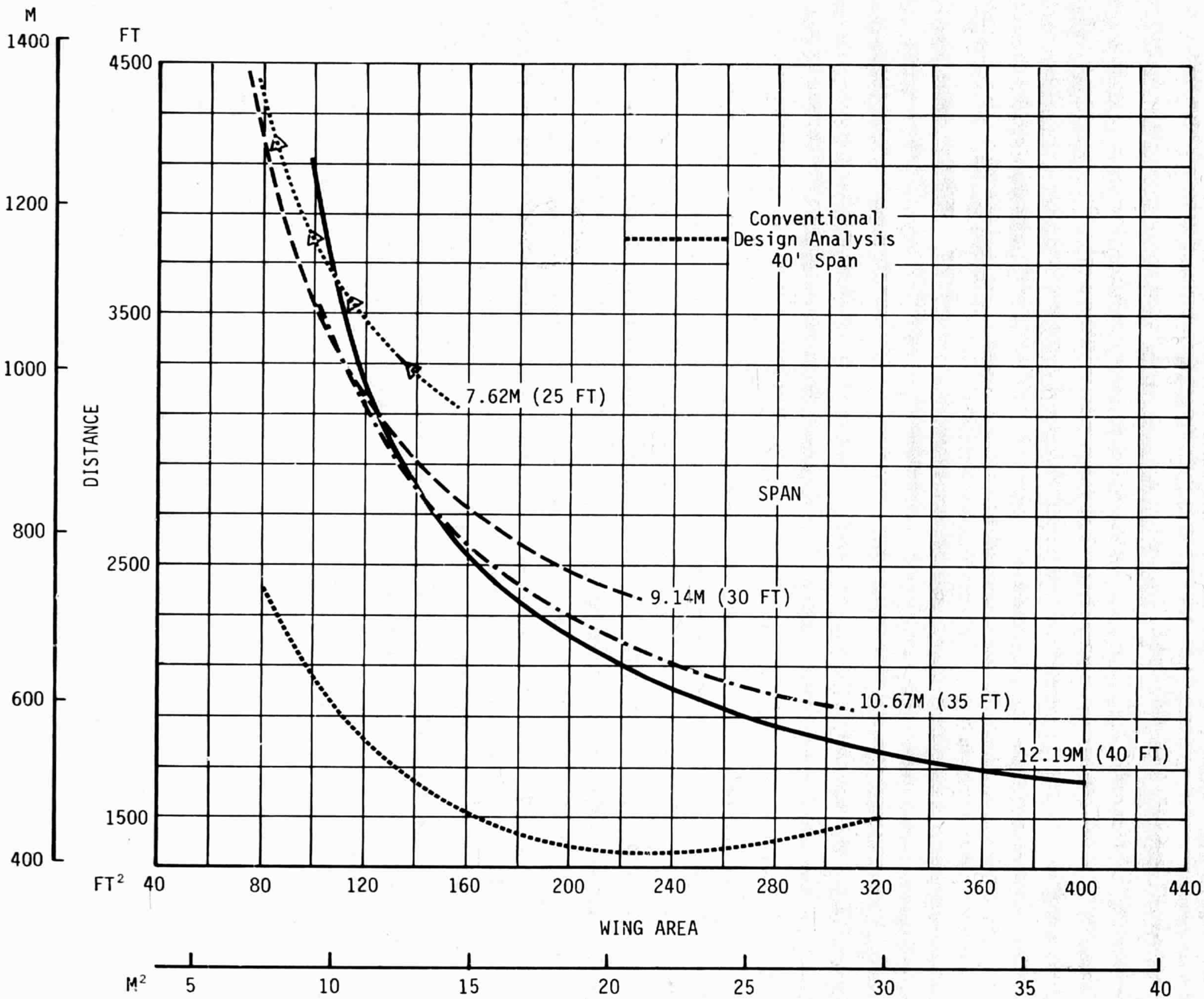

FIGURE 44 - TOTAL TAKEOFF DISTANCE OVER 15M (50 FT)

Retail cost is shown in Figure 55 from the GASP analysis. Explanations for the higher retail cost predicted by the conventional method are covered in detail in Section 6.9 and Appendix C.

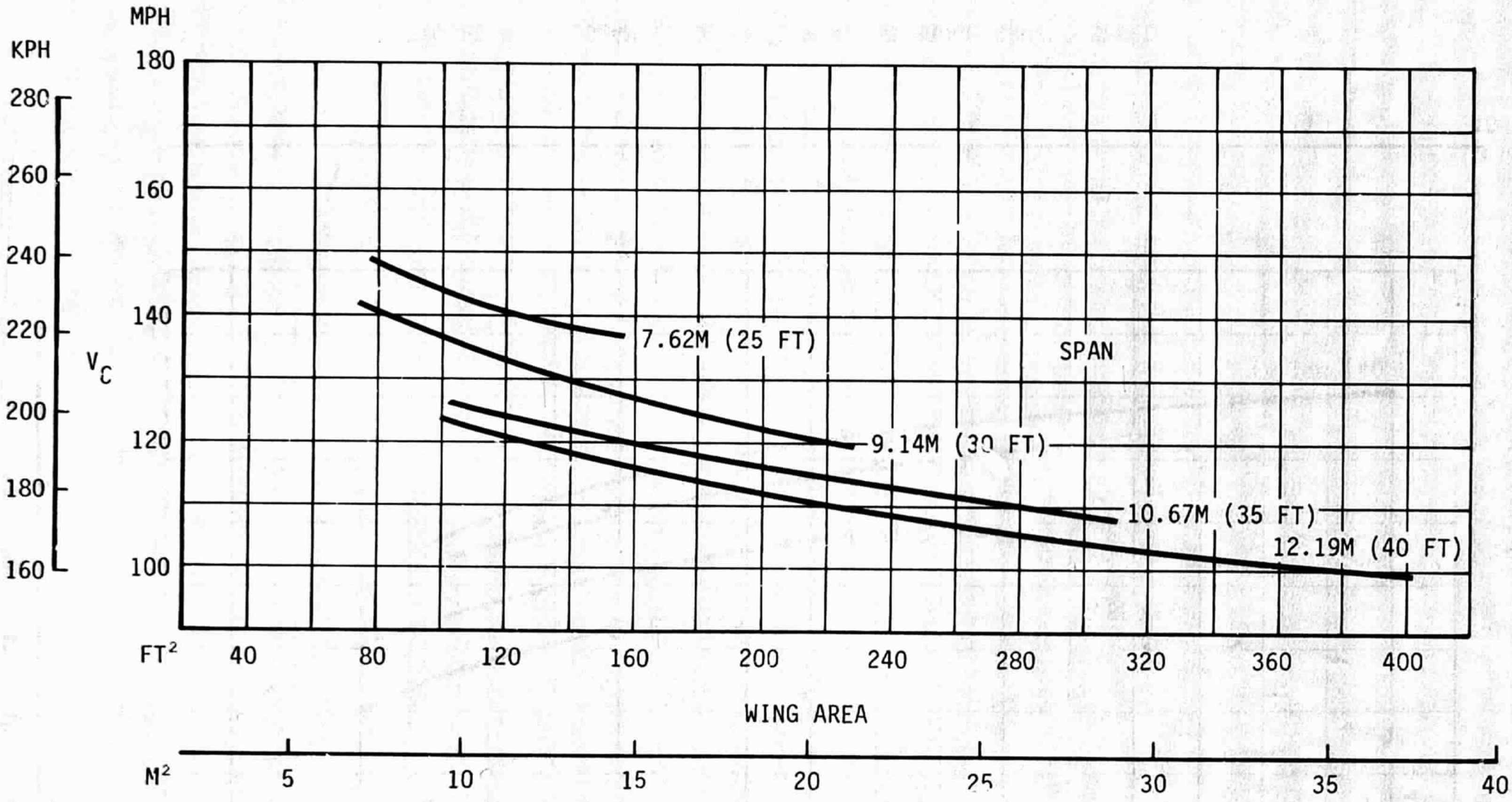

FIGURE 45 - SEA LEVEL MAXIMUM RANGE CRUISE SPEED

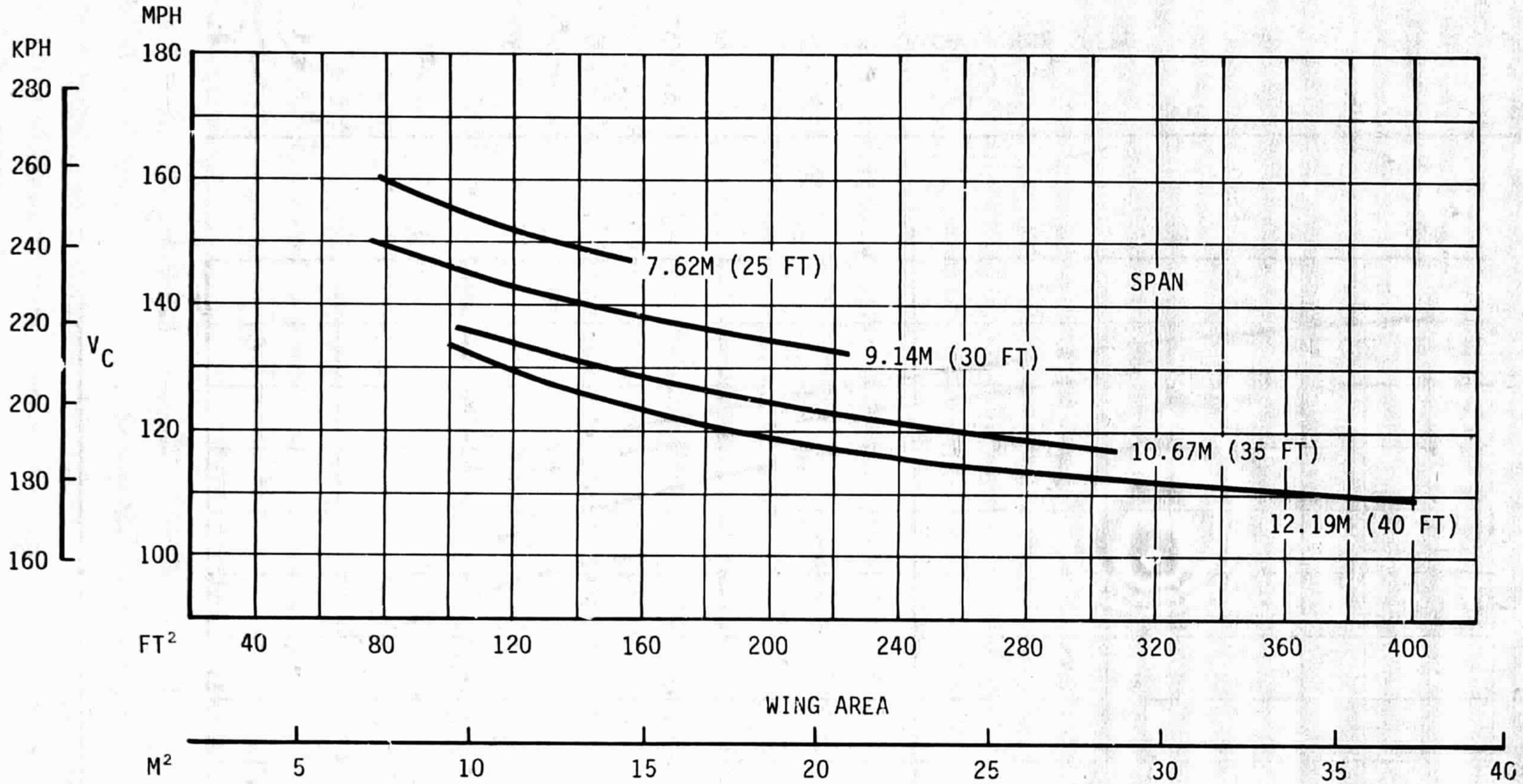

FIGURE 46 - 1524M (5,000 FT) MAXIMUM RANGE CRUISE SPEED

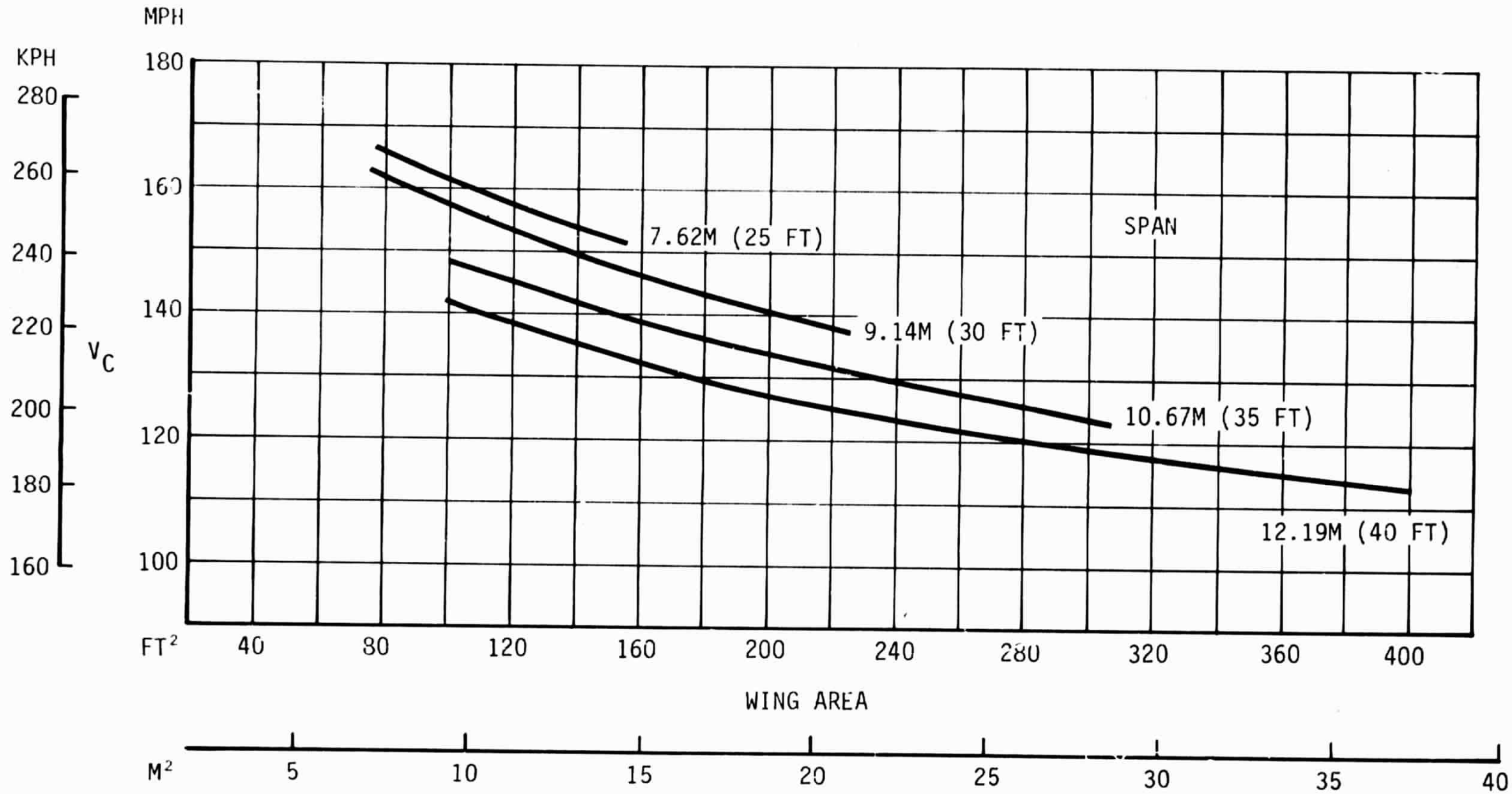

FIGURE 47 - 3048M (10,000 FT) MAXIMUM RANGE CRUISE SPEED

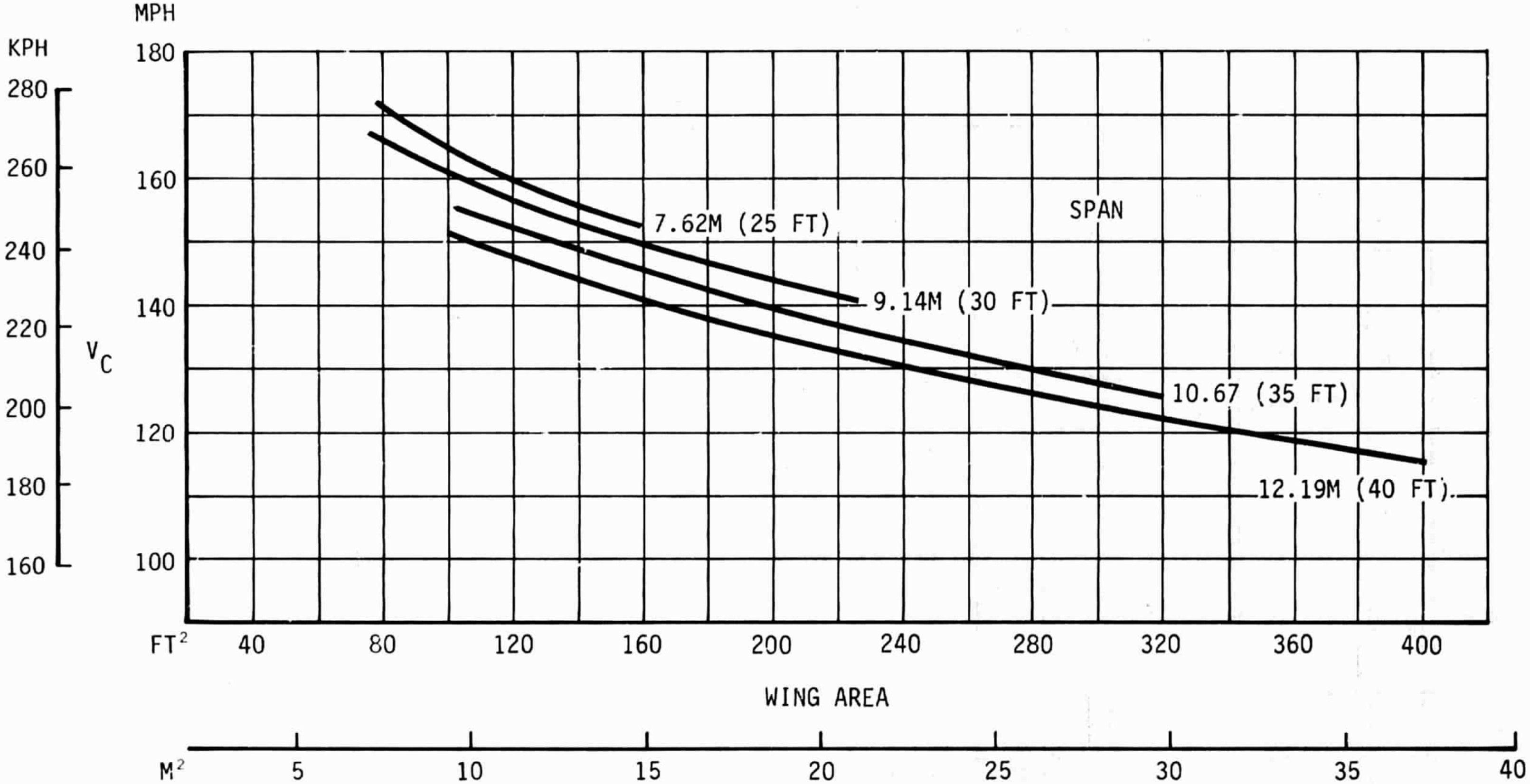

FIGURE 48 - 4572M (15,000 FT) MAXIMUM RANGE CRUISE SPEED

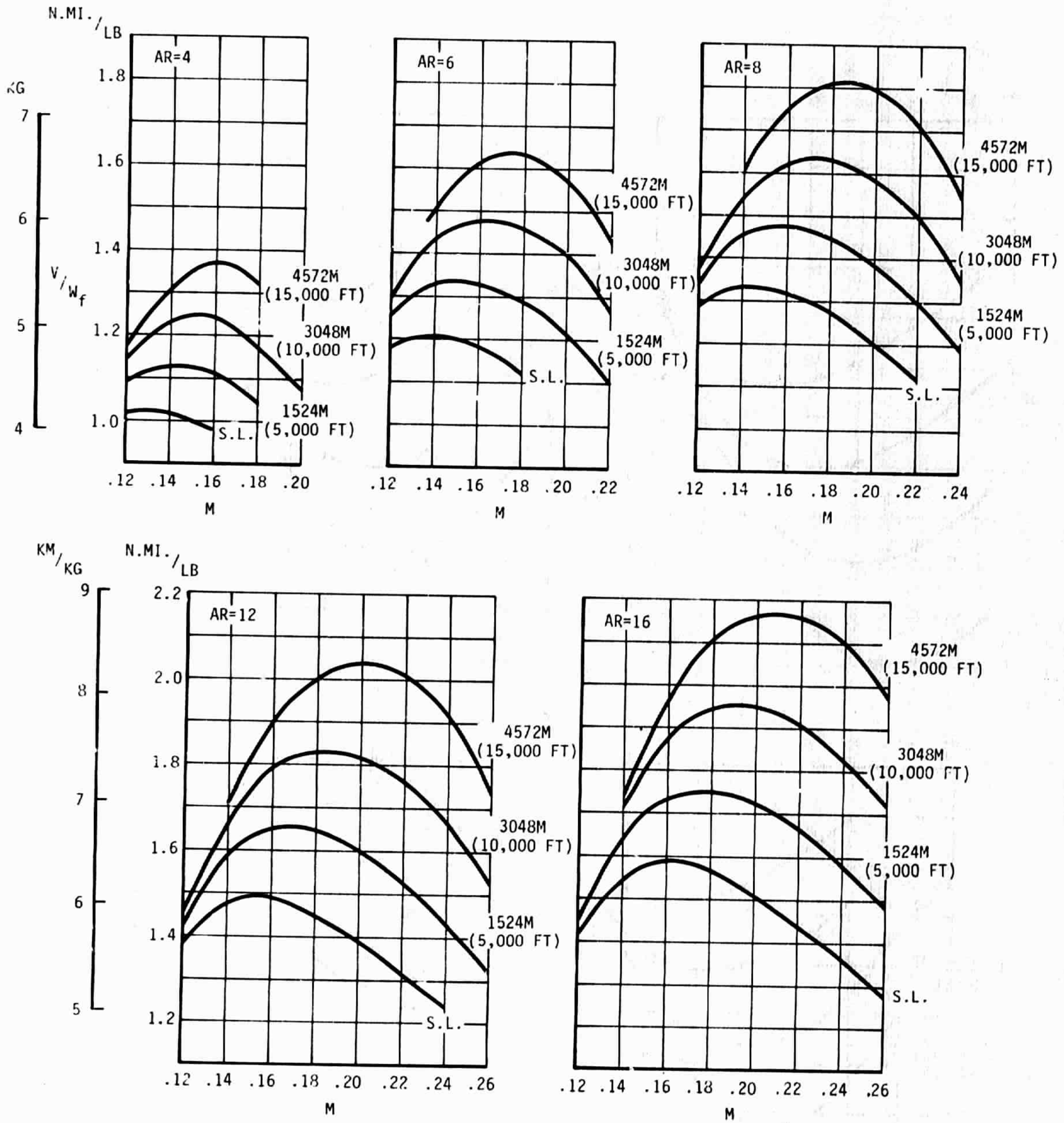

FIGURE 49 - RANGE FACTORS - 12.19M (40 FT) SPAN

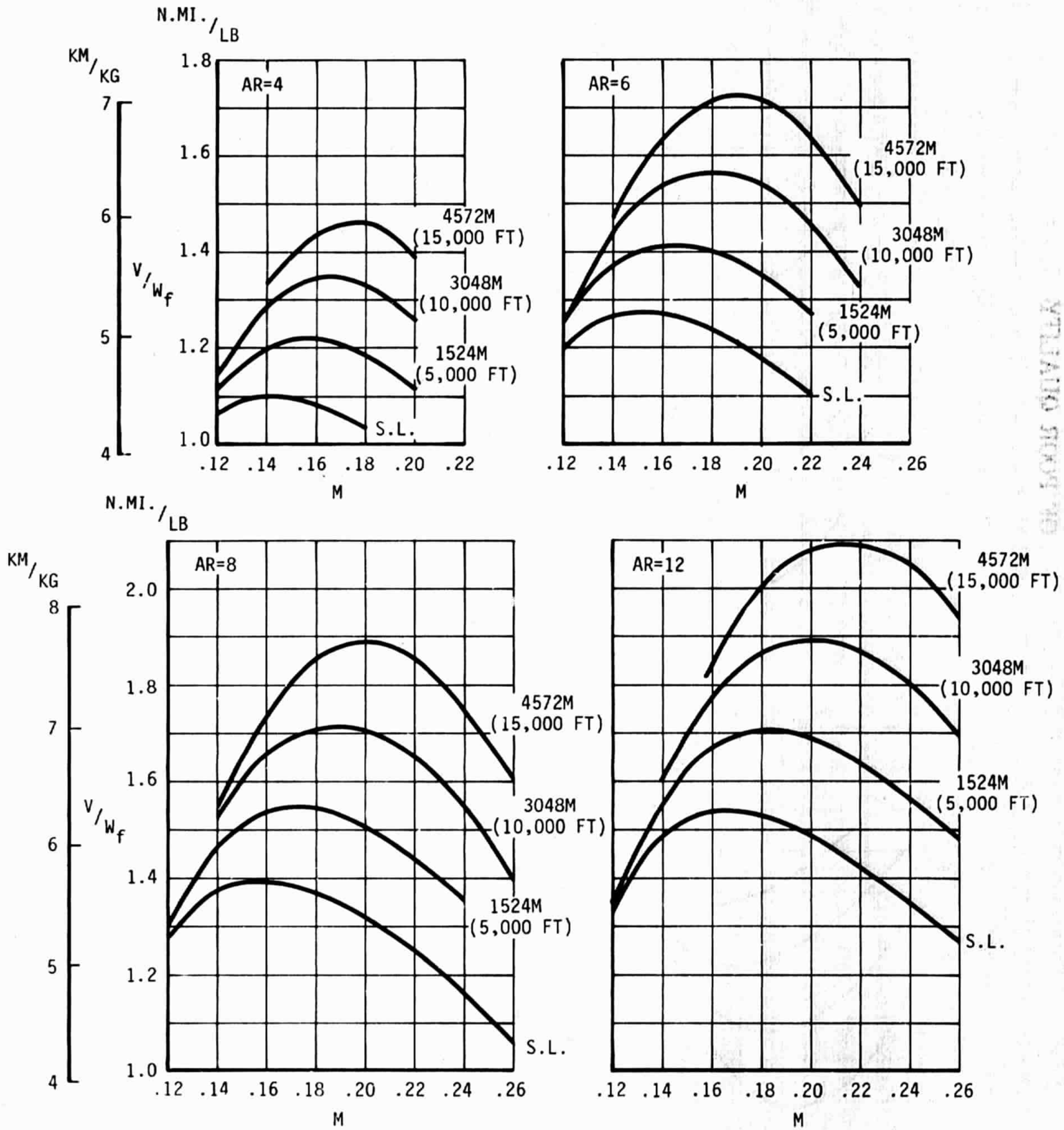

FIGURE 50 - RANGE FACTORS - 10.67M (35 FT) SPAN

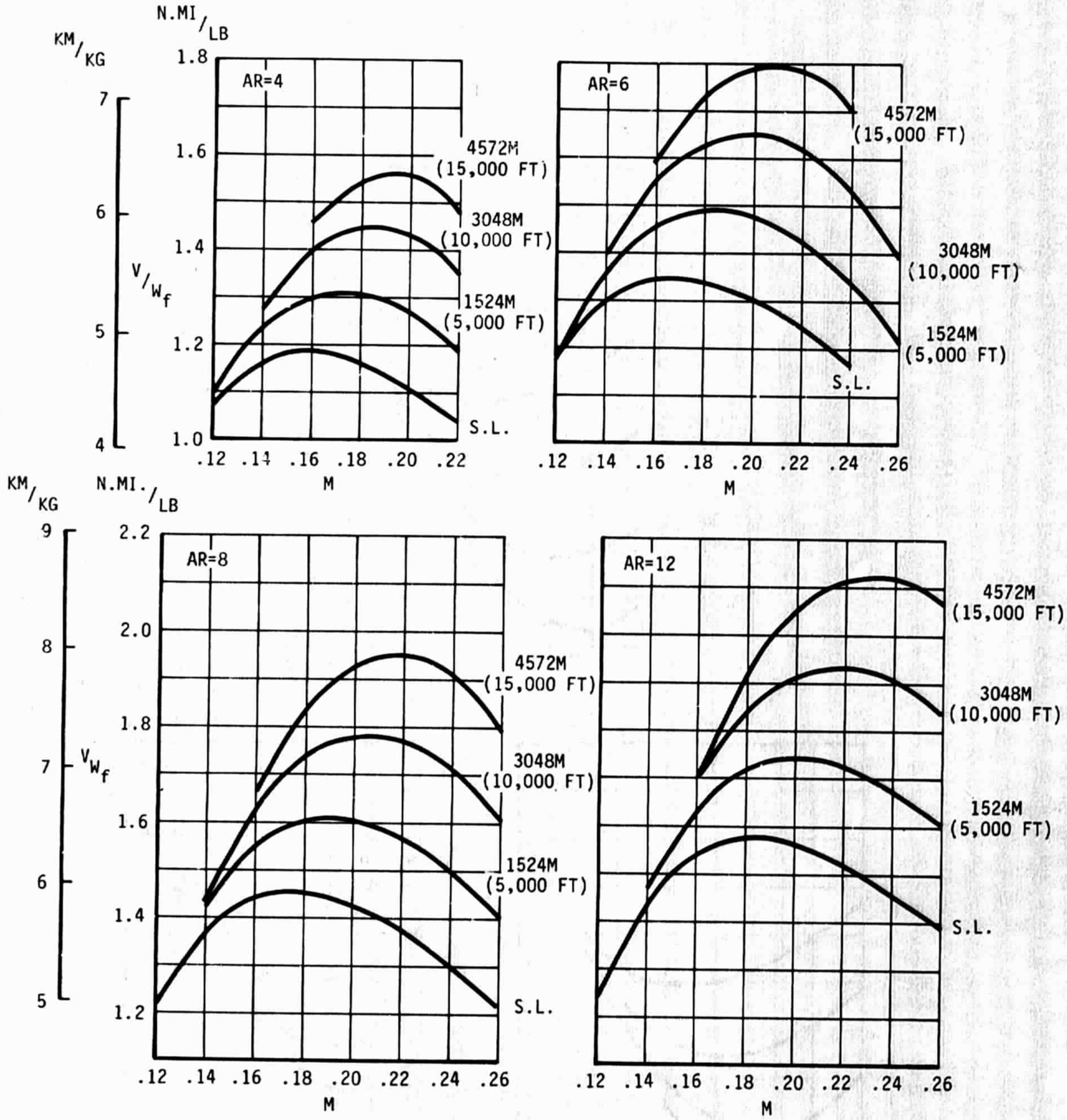

FIGURE 51 - RANGE FACTORS - 9.14M (30 FT) SPAN

ORIGINAL PAGE IS OF POOR QUALITY

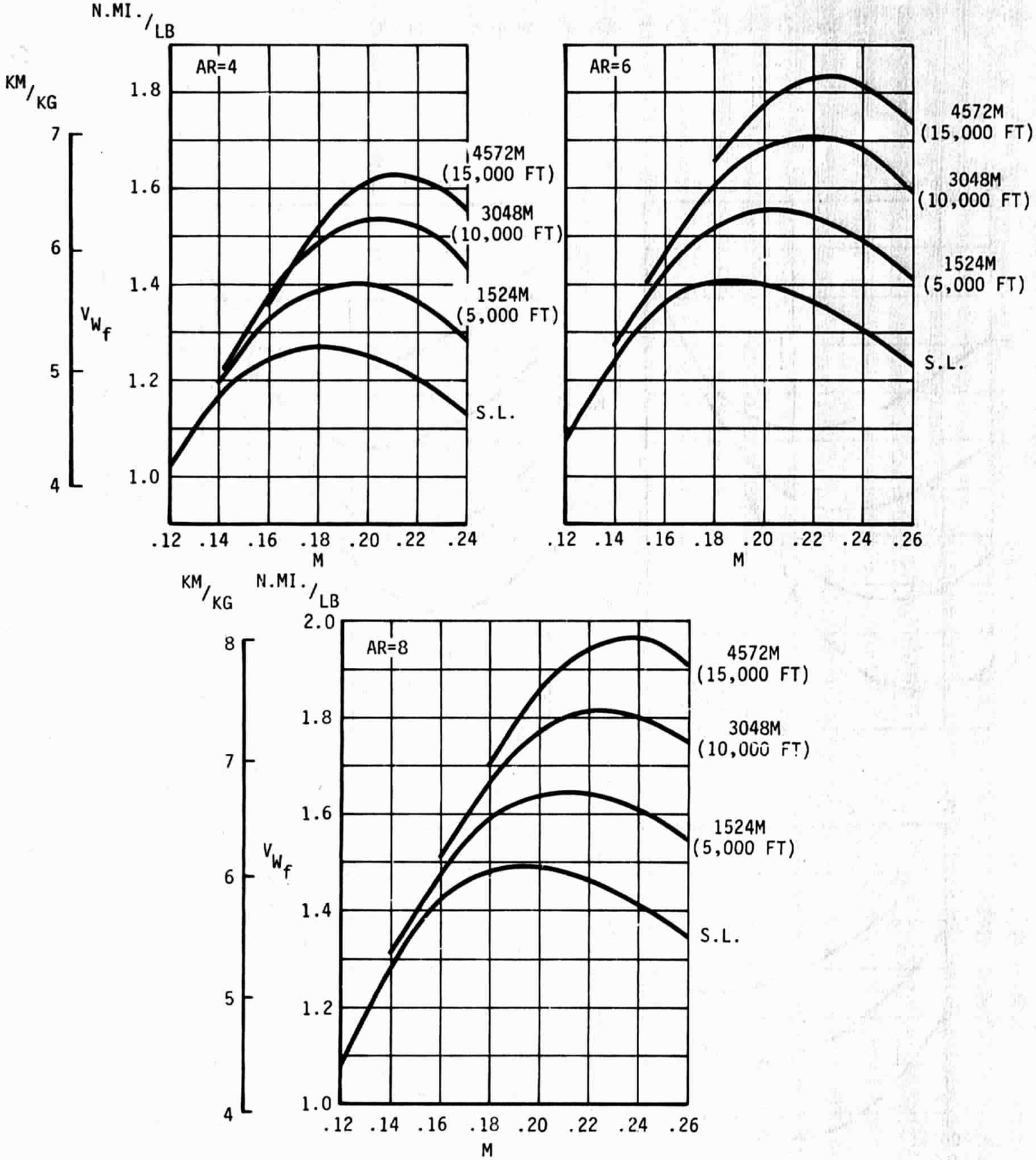

FIGURE 52 - RANGE FACTORS - 7.62M (25 FT) SPAN

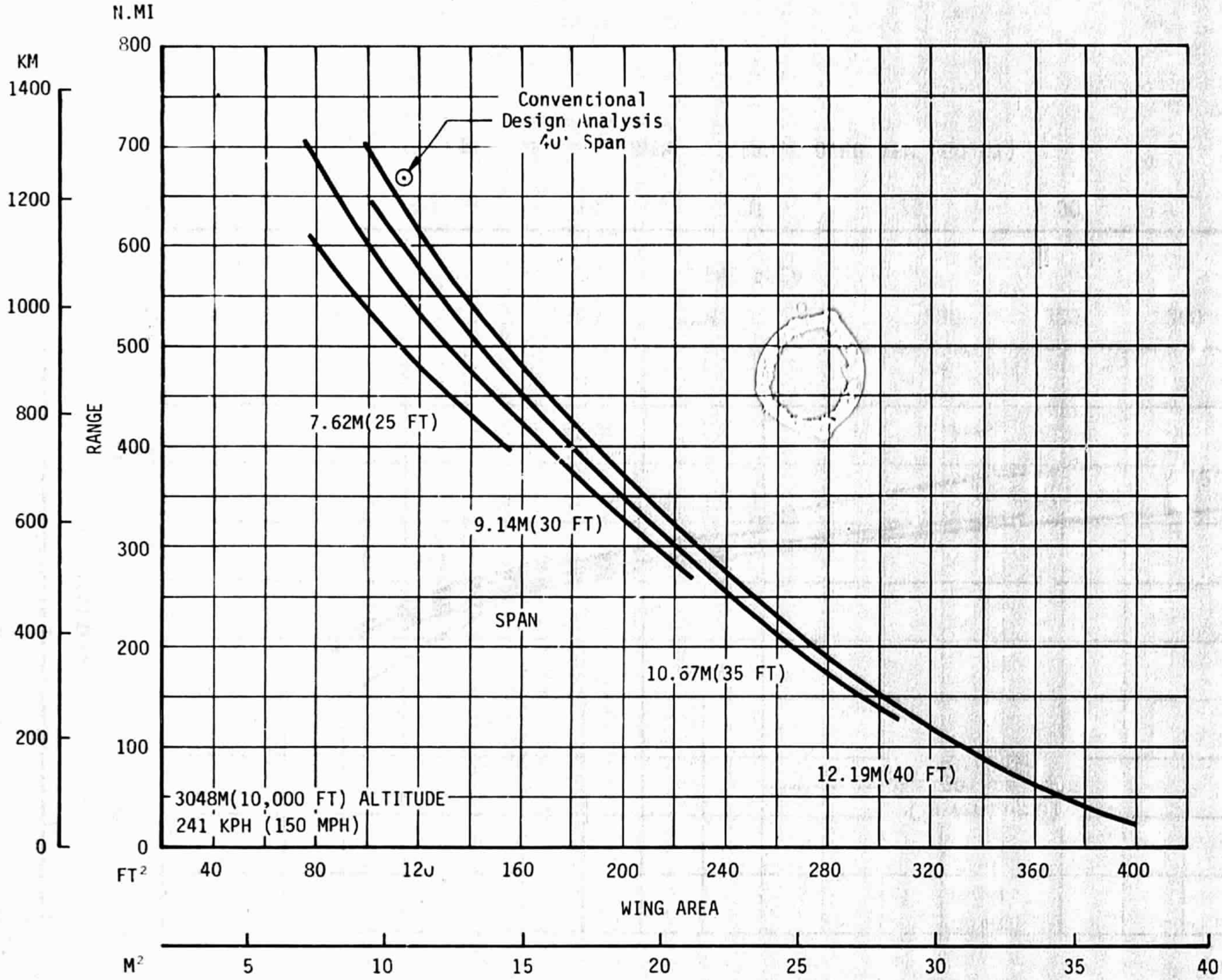

FIGURE 53 - RANGE

ORIGINAL PAGE IS OF POOR QUALITY

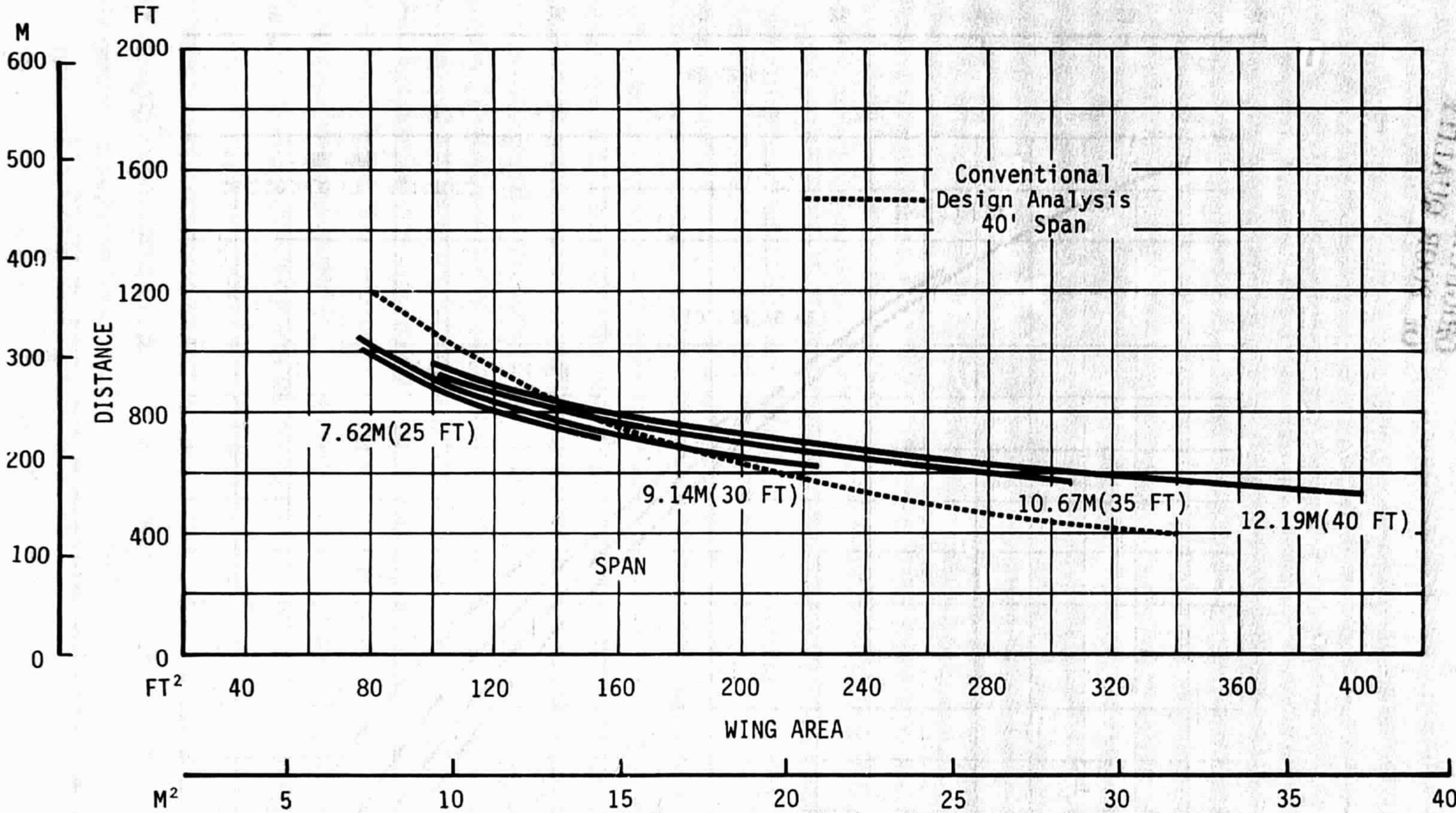

FIGURE 54 - LANDING DISTANCE OVER 15M (50 FT)

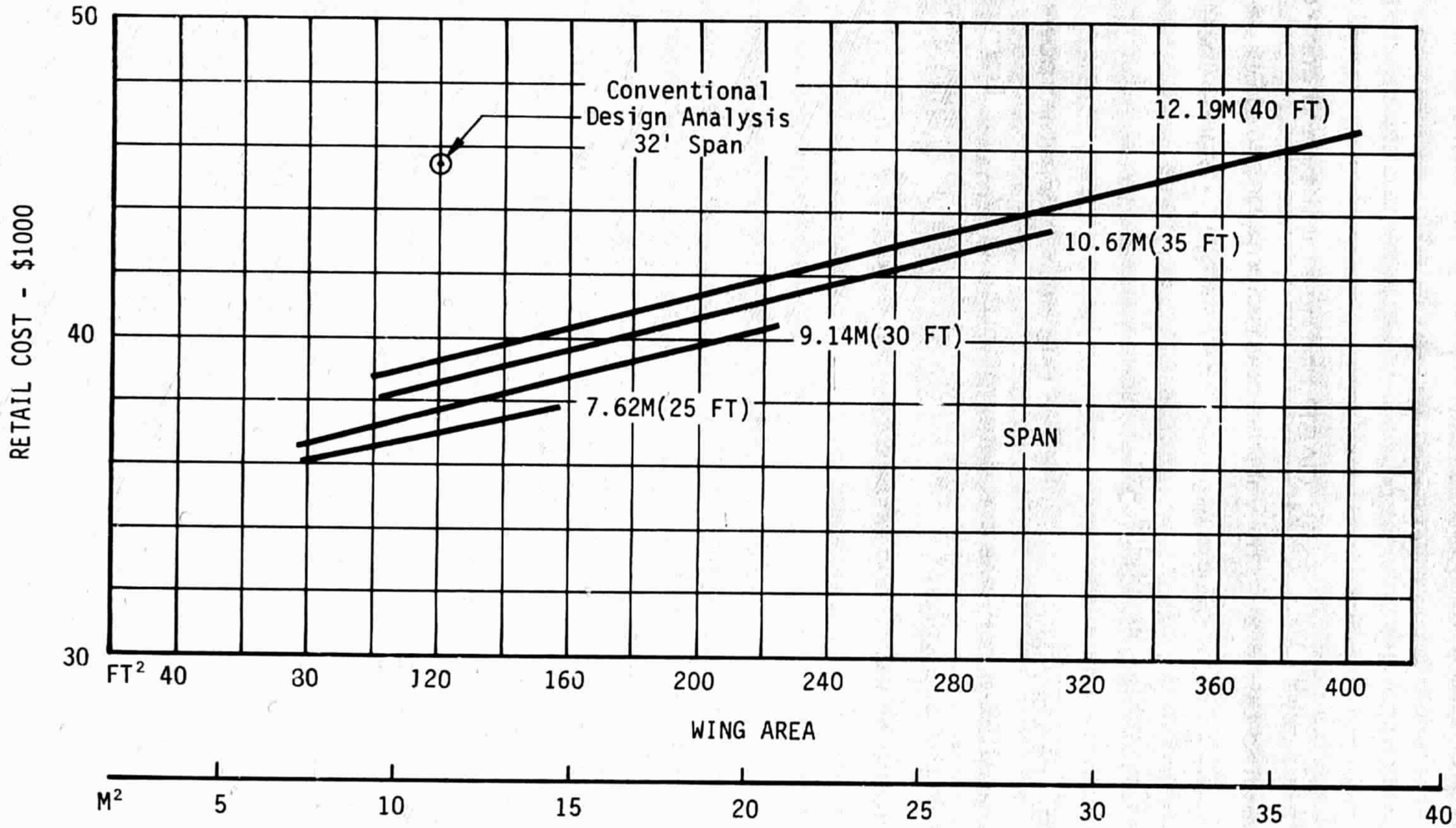

FIGURE 55 - COST

8.2 Items Covered by Conventional Process and Not by GASP

At the time the study was done, GASP had no stability and control analysis and handled tail sizing by volume coefficients only. This is a definite shortcoming of the GASP method. At present, volume coefficients must be computed externally and input into GASP.

In the GASP performance analysis for this study, constant power setting cruise was not available nor was the determination of V_{max}. Both of these are necessary in the analysis of a new airplane. Likewise noise calculations are not made in the analysis and should be included.

8.3 Items Covered by GASP and Not by the Conventional Process

The biggest single advantage of the GASP analysis procedure is that every design iteration meets all the requirements and constraints placed upon it. Therefore, every point that yields a solution is a potential design of the desired aircraft. This makes the process of optimization much simpler and quicker since every iteration exercises all of the disciplines included in the program. The conventional process cannot do this optimization in anything approaching the time required for GASP to handle the volume of iterations required.

8.4 Areas of Disagreement

In general, the methodologies are valid and the mathematic calculations are correct. The deficiencies occur in the limitations and assumptions.

For some of the default parameters, particularly the weight factors, there is some difficulty in selecting the values to be used in the program. After the wing weight factor was calculated by the conventional method for the PD 1502 airplane, the wing weight factor was cycled until the GASP calculated wing weight matched the hand calculated weight.

Other specific areas of disagreement have been found in the GASP program:

If the wing chord is reduced below 1/10 of the fuselage length, either by high aspect ratio or long fuselage, the program will not run.

There also appears to be some problem in the Part 23 rate of climb requirement. FAR 23.65 gives the takeoff climb requirement of 300 fpm or 11.5 V_{SI} (Kt) with takeoff flap and gear extended. The landing climb gradient requirement from FAR 23.77 is 200 fpm or 5.75 V_{SO} with takeoff power, landing flaps and gear extended.

8.5 Recommended Improvements to GASP

The GASP program is a large and complex routine and requires large computer storage capability. GLC was forced to divide the program into nine modules to allow using it with the IBM 370 system available. Similarly, it required GLC about two months of full time effort to get the program operational and to reproduce the check cases. Some of this time could have been saved if there were a comprehensive user's manual available. This manual is necessary not only for initial start up, but also for recurrent usage. This manual should include discussions and examples of the options available together with explanations of interaction between and among the various subroutines. It is presently very difficult to follow the logic flow as the various options are exercised.

In addition to the user's manual, a comprehensive technical documentation of the methods used is mandatory to evaluate the suitability of the GASP program for the particular application. The documentation should include the theories used and the assumptions employed in the analysis in order to allow recognition of the limitations inherent in the program.

Another possible improvement would be to provide optional logic flows for those cases where only a limited amount of data is desired or only certain disciplines need to be addressed. For example, it may be desired to study only the takeoff performance for a variety of designs. For this case it should not be necessary ot redefine the airplane and exercise the geometry, weight and sizing options for each point.

GASP is a useful analysis tool for preliminary design of a clean-sheet airplane. For derivative type aircraft, it is questionable whether the differences

from one design to another will be accurately modeled and evaluated. However, only further experience in using the GASP program in a working environment will determine its real value and limitations.

Appendix B gives a detailed discussion of the weight estimation methodology of GASP and some general comments on its logical flow.

9.0 TECHNOLOGY REQUIREMENTS

The major technology requirement for this class of aircraft is the manufacturing technology required to build turbine engines at prices competitive with piston engines. Turbine engines have certain advantages, both for the customer and the airframe manufacturer, therefore, the purchase price does not have to equal that of the piston engines. The price differential that the market is willing to support is unknown until some products are sold, but it is probably not more than 100%. In other words, if the turbine engine costs more than twice as much as a comparable piston engine, it probably will not be accepted in profitable volume in this market.

Light aircraft design and manufacture involves close attention to cost sensitive and weight sensitive factors that differ somewhat from larger aircraft. These factors include, but are not limited to:

A) Turbofan engine cycle, construction, weight, and cost.

B) Design simplification for low tooling and production costs.

C) A higher than normal sensitivity of aircraft weight to fixed equipment weight.

D) Aerodynamic configuration design for inherent stability without artificial stabilization and damping.

E) A high sensitivity to engine inlet efficiency.

F) A high sensitivity to engine placement in the airplane as it affects weight and balance, moment of inertia, and interference drag of the wing/fuselage/nacelle combination.

G) An optimum wing loading and aspect ratio for minimum purchase and operating cost.

H) A requirement for lower noise and emissions than for larger aircraft.

I) Freedom from ground support equipment requirements such as power carts, ladders, work stands, etc.

These factors have been considered to varying degrees of detail in this study. All of them have been, or can be, resolved in a satisfactory manner. Economics aside, present technology is adequate to physically build a four place turbofan powered light airplane. Engines and all necessary airframe equipment and materials required to build a good light airplane are available today.

Unfortunately, economics cannot be set aside. Economics is the primary reason for the existence of small airplanes as well as the companies which build them. Were it not for economics, we would all fly large comfortable high performance airplanes.

Although technology advances are not required to build a turbofan powered light plane, they would be beneficial, as in any branch of commerce. In this cost sensitive industry, however, the benefit of a particular item depends on its cost factors. For instance, advanced composites will see little use until the material costs come down, since this is a production cost. Conversely, the only cost connected with an advanced airfoil is a possible slight increase in development cost. High lift device costs are variable; contour modifications are free, but addition of elements or power is expensive. Therefore research and development is useful and desirable and should be continued, but the effect of each item on manufacturing cost must be carefully considered.

10. CONCLUSIONS AND RECOMMENDATIONS

As a result of this study, the use of a program such as GASP has been judged to be of significant value as an advanced design tool. The GASP program itself, due to its broad scope of coverage, would be particularly useful in this role if the following features were altered:

Documentation - Probably the most seriously lacking element of the program is the detail information and methodology of the program operation and subroutine computation. A typical example of the type of problem that results from this is that when some program options are exercised, unwanted sizing occurs without an apparent method to force the program back to a baseline configuration.

Simplification - The original GASP program represented the efforts of many programmers who contributed subroutines and modules to the overall program makeup. Since this program is operational, much could be done by a single programmer, with an overall view, to streamline the data flow, simplify the input, reduce computational time, reduce core requirements and minimize initial loading problems.

Flexibility - In its current form, parametric studies require repeated program submittals to obtain sensitivity factors on geometry or performance requirements. The ability to stop the computational process at a given point and perform parametrics on a particular independent variable would be desirable. Examples of this would be the effect of wing geometry on cost or cruise speed on range.

Improvement in accuracy - Overall prediction of performance, weight and costs of some classes of aircraft is good. For the present study case, several areas presented data either inappropriate or out of date; detailed descriptions of the discrepancies are outlined in the text or appendicies. In general, the aerodynamic data generated by GASP adhered closest to the results of the contempoerary design methods with weight and cost predictions being somewhat more at odds. The following are some specific areas of disagreement:

a) Takeoff and landing distances - While parametric trends predicted by GASP and contemporary methods were similar, there were significant differences in the air and ground distances calculated by the two methods. These discrepancies were not resolved during the study and are the apparent results of differences in methodology which should be investigated.

b) Wing weight prediction - As pointed out in the critique of the WGHT module in GASP (Appendix B), the default values in GASP are inappropriate for this class of aircraft and the weight variation with geometry predicted by GASP exceeds that obtained by contemporary prediction methods. It is recommended that for better applicability to this class of aircraft, more statistical data be added to the GASP methodology, particularly at higher aspect ratios, and the default values adjusted accordingly.

c) Cost prediction - Appendix D presents a detailed critique of the GASP cost module along with comparisons to contemporary cost figures for corresponding cost elements. It is recommended that as a minimum this module should be updated to account for more current costs and accounting practices. Ideally, the methodology should be based on the commonly accepted AMPR weight concept and discrete inputs should be provided to allow direct adjustment to items such as material costs, manpower rates and learning curve improvement.

Expansion of program capability - Currently, some desireable information is not available from the GASP data or is obtainable only by repeated submittals or time consuming cross plotting. It is recommended that the following additional capabilities be added to the GASP progaram:

a) The option of performing a stability and control analysis.

b) The ability to calculate V_{max}.

c) The capability of performing a constant power setting cruise.

REFERENCES

1. Merrill, G. L.: Study of Small Turbofan Engines Applicable to Single Engine Light Airplanes. NASA CR-137944, AiResearch Manufacturing Company of Arizona, 1976.

2. McGhee, J.R. and Beasley, W.D.: Low Speed Aerodynamic Characteristics of a 17 Percent Thick Airfoil Section Designed for General Aviation Applications. NASA TN D-7428, 1973.

3. Wentz, W.H., Jr. and Seetharam, H.C.: Development of a Fowler Flap System for a High Performance General Aviation Airfoil. NASA CR-2443, 1974.

4. Wentz, W.H., Jr. and Volk, C.G., Jr.: Reflection Plane Tests of Spoilers on an Advanced Technology Wing with a Large Fowler Flap. NASA CR-2696, 1976.

5. Perkins, C.D. and Hage, R.E.: Airplane Performance, Stability and Control. John Wiley and Sons, Inc., 1953.

6. Purser, P.E. and Campbell, J.P.: Experimental Verification of a Simplified Vee Tail Theory and Analysis of Available Data on Complete Models with Vee Tails. NACA TR 823, 1945.

APPENDIX A - SAMPLE GASP RUN

Run 21, contained in this appendix is the second baseline run of the study and represents a standard synthesis type run.

```
+++++START OF INPUT FOR CONTROL
      / FOUR SEAT UTILITY AIRCRAFT GLC MODEL1502    GARRETT ENGINE
      NPC#1,       NSC#1,      NTYP#10,  EMCRU#.2041,          HNCRU#10000.,
      EM#.2041,  ALT#10000.,ENP#1.,   WG#2000.,  WGS#25.0, KARITE#2,
      *

+++++START OF INPUT FOR VEHICLE SIZING
      SAH#2.,      AS#20.,     VAS#0.,     PAX#2.,    HCK#1.833,AS#1.,
      ESW#0.,      ELDCN#1.567,            ELDC#9.5, ELDCT#1.96,
      DELP#0.,     KCONFG#0, AR#12.,      TCR#.17,
      TCT#.17,     SLM#1.,    DLPC#0.,   YP#0.,     YNG#0.,
      EYE#0.,      ALPHLW#-3.4,           IGEAR#1,   CATD#1,
      SAH#0.,      ARH#5.26,ARVT#1.10,TCHT#.10,
      TCVT#.10,    SLH#.5,    SLVV#.5,VBARHX#1.11,
      VBARVX#.078,            CDELTH#.221,          BOELTV#2.65,
      WFESL#200.,             ELR#0.,   ELN#5.0,
      NSC#1,
      WFUL#200.,  AFEX#214.,
      *

+++++START OF INPUT FOR CONTROL
      NPC#2,NSC#1,*

+++++START OF INPUT FOR CONTROL
      NPC#1,NSC#7,*

+++++START OF INPUT FOR CRUISE AERODYNAMICS
      CK#2.,      CKF#1.313,CKN#.33,  CKVT#0.,  CKHT#2.67,
      CKTP#0.,    GFEE#.2087,DELFE#0.,  DELCD#0.,*

+++++START OF INPUT FOR CONTROL
      NPC#2,NSC#2,*

SUMMARY OF CRUISE LIFT-WEIGHT BALANCE
  ANGLE OF ATTACK(DEGREES)      2.329 LIFT     2000.0 L/D     17.475 ALTITUDE    10000.0

+++++START OF INPUT FOR CONTROL
      NPC#1,NSC#14,*

+++++START OF INPUT FOR FLAP PARAMETERS
      JFLTYP#6,   VKTIN#80.,CFOC#.3,
      DELFD#10., DELLED#0.,PCLMAX#1.41,BTEOB#1.0,
      IFLAP#1,
      *
    DCLMTE,DELTEO,DCDOTE,WCFLAP =    1.6700  30.0000    0.1000    0.6330

+++++START OF INPUT FOR CONTROL
      NPC#2,NSC#14,*

TAKEOFF**  DELCL=  0.9677  DELCD=  0.0219  CLMAX=  2.3921  DELTA6=  0.9900  SIGMTO =  0.285

+++++START OF INPUT FOR CONTROL
      NPC#1,NSC#14,*

+++++START OF INPUT FOR FLAP PARAMETERS
      IFLAP#2,    DELFD#40.,
      *
    DCLMTE,DELTEO,DCDOTE,WCFLAP =    1.6700  30.0000    0.1000    0.6330

+++++START OF INPUT FOR CONTROL
      NPC#2,NSC#14,*

LANDING**  DELCL=  2.3350  DELCD=  0.1484  CLMAX=  3.7595  DELTA6=  0.8550  SIGMLD =  0.580

+++++START OF INPUT FOR CONTROL
      NPC#1,NSC#12,*

+++++START OF INPUT FOR LANDING MANUVER
      XLGGRO#2000.,
      *

+++++START OF INPUT FOR CONTROL
      NPC#2,NSC#12,*

   ****************************************************************************************

               TEMP=  518. DEGISTD+ 0.
```

```
LANDING WING LOADING=  25.00 PSF.
LANDING WEIGHT =      2000. LBS.

LANDING DISTANCE FROM 50. FT.=   1013. FT.

F.A.R. FACTORED FIELD LENGTH =   1689. FT.

                 APPROACH                 TRANSITION              DELAY                 ROLL
              DIST=        553.     DIST=          29.    DIST=         86.    DIST=          345.
              R/S=         568.     XLFNX=       1.200    TDELAY=      1.00    MUB=         0.3500
              VAPEAS=     57.51     SINKID=      3.000    TIDLE=        0.     TR/TIDLE= 0.0
              VAPTAS=     57.54     VSTLAS=      44.24                         ABAR(G)=   0.3326
              THETA=       5.16     CLMX=       3.7666
              THRUST=        0.
 IDLE THRUST LIMITING RATE OF SINK

 *******************************************************************************************

+++++START OF INPUT FOR CONTROL
     NPC=1,NSC=6,*

+++++START OF INPUT FOR MISSION DESCRIPTION
     ICOND=1,ISEG=9,MTG=2,17,THEMAX=6.5,
     *

+++++START OF INPUT FOR CONTROL
     NPC=1, NSC=10, KWRITE=-1,*

+++++START OF INPUT FOR PROPULSION SIZING
     XTORQ=2000.,JENGSZ=1,NCCPU=300.,
     MITR=.5232, SMID=.35, ALUDE=3.53,  PR=.995,
     *

+++++START OF INPUT FOR CONTROL
     NPC=2,NSC=10,*

VSTLKT=  55.4 KTS EAS    VRAT= 1.200 CLTO= 1.6661
     VEND =  113.9 KNOTS EAS

ROTATION (TIME=   23.7 AND TAS=     65.4 EAS=     65.4)
LIFTOFF (TIME=   25.8 DIST=    1629.7 TAS=     69.9 EAS=     69.9)
DISTANCE TO 35 FT.=  2223.1

 ITERATION TO MATCH TAKEOFF DISTANCE
  XTO,XTORQ,WASLS  2223.  2000.   24.42

VSTLKT=  55.4 KTS EAS    VRAT= 1.200 CLTO= 1.6661
     VEND =  116.9 KNOTS EAS

ROTATION (TIME=   21.2 AND TAS=     65.4 EAS=     65.4)
LIFTOFF (TIME=   23.2 DIST=    1470.6 TAS=     70.3 EAS=     70.4)
DISTANCE TO 35 FT.=  2024.1

 ITERATION TO MATCH TAKEOFF DISTANCE
  XTO,XTORQ,WASLS  2024.  2000.   26.87

VSTLKT=  55.4 KTS EAS    VRAT= 1.200 CLTO= 1.6661
     VEND =  119.8 KNOTS EAS

ROTATION (TIME=   20.9 AND TAS=     65.4 EAS=     65.4)
LIFTOFF (TIME=   22.8 DIST=    1439.2 TAS=     70.1 EAS=     70.1)
DISTANCE TO 35 FT.=  2004.6

 ITERATION TO MATCH TAKEOFF DISTANCE
  XTO,XTORQ,WASLS  2005.  2000.   27.16

ENGINE SIZED TO MATCH CRUISE DRAG - SLS AIRFLOW=      24.42

ENGINE SIZED TO MATCH TAKE OFF DISTANCE OF 2000.0FEET - SLS AIRFLOW=      27.16

RATED SEA LEVEL STATIC THRUST PER ENGINE=     425.8 LBS
 *******************************************************************************************

+++++START OF INPUT FOR CONTROL
     NPC=1, NSC=13, KWRITE=2,*

+++++START OF INPUT FOR PROPULSION WEIGHT
     [illegible]
```

ORIGINAL PAGE IS OF POOR QUALITY

```
*****START OF INPUT FOR CONTROL
     NPC#2,NSC#13,*

PROPULSION SYSTEM WEIGHTS
  ENGINE WEIGHT/ENGINE                     85.2
  NACELLE WEIGHT/ENGINE                    24.3
  PYLON WEIGHT/ENGINE                       0.0
     PROP OR OFAN                  0.0
     GEARBOX                       0.0
     SHROUD                        0.0

ENGINE POD DIMENSIONS
  ENGINE FACE DIAMETER(FT)              1.43
  NACELLE LENGTH(FT)                    5.03

*****START OF INPUT FOR CONTROL
     NPC#1, NSC#4, KWRITE#2,*

*****START OF INPUT FOR VEHICLE WEIGHT OR WING LOC.
     SKLG#.054,SKW##110.,SKB# 85.,SKY#.16,SKZ#0.,
     SKFT#0.,LCWING#1, RELP#.658,RFLR#.36,DWPOCH# 27.,DWPOCV#32.,
     SKW##.95, SKFS#.06, SKF##.45,
     *

*****START OF INPUT FOR CONTROL
     NPC#2,NSC#4,*

SUMMARY OF CRUISE LIFT-WEIGHT BALANCE
  ANGLE OF ATTACK(DEGREES)      2.329 LIFT     2000.0 L/D     17.105 ALTITUDE   10000.0

 WING LOCATION INFO.
FUSELAGE LENGTH        =    21.91          H-TAIL VOL. ARM        =     9.79          C.G.LOCATION OF PROPULSION=    14.42
WING 1/4C LOC.ON C.L.=      8.82          H-TAIL C.G.LOCATION    =    19.11          C.G.OF REMAINING WEIGHT   =     7.89
MAC 1/4C LOCATION      =     8.99          H-TAIL MAC FROM C.L. =     2.45
MAC DIST.FROM C.L.     =     7.75          H-TAIL LOCAT ON VERT.=     0.0
WING C.G.LOCATION      =     9.25          V-TAIL VOL. ARM        =    10.39
TIP TANKS C.G.LOCATE =       0.0           V-TAIL C.G.LOCATION    =    19.72

 AIRCRAFT C.G. LOCATION =     8.99 FT. OR 0.250 OF MAC

                          WING          H-TAIL          V-TAIL
  AREA              [illegible]0.000          23.423          18.603
  SPAN                  30.984          11.100           4.524
  ASPECT RATIO          12.000           5.260           1.100
  TAPER RATIO            1.000           0.500           0.500
  1/4C. SWEEP            0.0            27.000          32.000
  L.E. SWEEP             0.0            29.808          42.858
  C.L. CHORD             2.582           2.818           5.483
  MEAN CHORD             2.582           2.188           4.265
  TIP CHORD              2.582           1.407           2.742

*****START OF INPUT FOR CONTROL
     NPC#2,NSC#5,*
```

GROSS WEIGHT = 2000. PASSENGERS = 2. PLUS CREW OF 1

FUSELAGE	LENGTH	(ELF)	21.91	FT
	WIDTH	(SWF)	4.33	FT
	WETTED AREA	(SF)	223.	SQFT
	DELTA P	(DELP)	0.0	PSI
WING	ASPECT RATIO	(AR)	12.00	
	AREA	(SW)	80.0	SQFT
	SPAN	(B)	31.0	FT
	GEOM. MEAN CHORD	(CHARW)	2.58	FT
	QUARTER CHORD SWEEP	(DLMC4)	0.0	DEG
	TAPER RATIO	(SLM)	1.000	
	ROOT THICKNESS	(TCR)	0.170	
	TIP THICKNESS	(TCT)	0.170	
	WING LOADING	(WGS)	25.0	PSF
	WING FUEL VOLUME	(VFW)	22.2	CUFT
HOR. TAIL	ASPECT RATIO	(ARHT)	5.26	
	AREA	(SHT)	23.4	SQFT
	SPAN	(BHT)	11.10	FT
	MEAN CHORD	(CBARHT)	2.19	FT
	THICKNESS/CHORD	(TCHT)	0.100	
	MOMENT ARM	(ELTH)	9.8	FT
	VOLUME COEFF.	(VBARH)	1.110	
VERT. TAIL	ASPECT RATIO	(ARVT)	1.10	
	AREA	(SVT)	18.6	SQFT
	SPAN	(BVT)	4.52	FT
	MEAN CHORD	(CBARVT)	4.26	FT
	THICKNESS/CHORD	(TCVT)	0.100	
	MOMENT ARM	(ELTV)	10.4	FT
	VOLUME COEFF.	(VBARV)	0.078	
ENG.NACELLES	LENGTH	(FLN)	5.03	FT
	MEAN DIAMETER	(DBARN)	1.43	FT
	NUMBER ENGINES	(ENP)	1.0	
	WETTED AREA	(SN)	22.51	SQFT

```
VDIVE =  241. KTS    VMO =  205. KTS    MMO = 0.616
ULT. LF =  5.70    MAN. LF =  3.80    GUST LF =  3.70

PROPULSION GROUP
    PRIMARY ENGINES                (WEP)           85.
    PRIMARY ENGINE INSTL.          (WPEI)          11.
    FUEL SYSTEM                    (WFSS)          27.
    PROPULSOR WEIGHT               (WPROP)          0.
    TOTAL PROP.GROUP WT.           (WP)           123.

STRUCTURES GROUP
    WING                           (WW)           218.
    HOR. TAIL                      (WHT)           42.
    VERT. TAIL                     (WVT)            0.
    FUSELAGE                       (WB)           181.
    LANDING GEAR                   (WLG)          108.
    PRIMARY ENG. SECTION           (WPES)          2A.
    GROUP WEIGHT INC.              (DELAST)         0.
    TOTAL STRUC.GROUP WT.          (WST)          573.

FLIGHT CONTROLS GROP
    COCPIT CONTROLS                (WCC)           15.
    FIXED WING CONTROLS            (WCFW)          28.
    SAS                            (WSAS)           0.
    GROUP WEIGHT INC.              (DELWFC)         0.
    TOTAL CONTROL WT.              (WFC)           42.

WT. OF FIXED EQUIPMENT             (WFE)          214.

WEIGHT EMPTY                       (WE)           953.

FIXED USEFUL LOAD                  (WFUL)         200. (INC. CREW OF 1)

OPERATING WEIGHT EMPTY             (OWE)         1153.

PAYLOAD                            (WPL)          400. (PAX=  2.)

FUEL                               (WFA)          447. (WFW=  447.) (WFTP=     0.)

GROSS WEIGHT                       (WG)          2000.
```

```
CRUISE MACH = 0.204          CRUISE ALTITUDE =  10000.

CRUISE RE.NUM. PER FT.    = 1.094E 06    FLATPLATE CF AT RE=10EX7 IS 0.00291

AERODYNAMIC DATA

                           FLATPLATE                  WETTED
DRAG BREAKDOWN             AREA(SQFT)      CDO       AREA(SQFT)

       WING                 0.5756      0.00720        148.81
       FUSELAGE             0.7413      0.00927        222.65
       VERT.TAIL            0.0         0.0             37.21
       HOR. TAIL            0.2316      0.00290         46.85
       ENGINE NAC.          0.0239      0.00030         22.51
       TIP TANKS            0.0         0.0              0.0
       INCREMENTAL          0.0         0.0              0.0
       FIXED GEAR           0.2087      0.00261      NOT INCL.

          TOTAL             1.7811      0.02226        478.03

          MEAN SKIN FRICTION COEF.= 0.003726

AERODYNAMIC COEFF.
       A1                                          0.6832
       A2                                         -0.1155
       A3                                          0.0783
       A4=.75X(T/C)                                0.1275
       A5=CDO**                                    0.0148
       A6                                          2.4742
       A7=1/(PI.SEE.AR)                            0.0351
       3-D LIFT SLOPE AT CRUISE MACH    (CLALPH)   5.4179    PER RADIAN
       OSWALD FACTOR                    (SEE)      0.7562

CRUISE  CD = 0.0223 +  0.0351 CL**2

+++++START OF INPUT FOR CONTROL
      NPC=2,NSC=6,*

+++++START OF INPUT FOR MISSION DESCRIPTION
      ISEG=1,ICOND=0,DELTT=.1833,*
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OF POOR QUALITY

MISSION PERFORMANCE DATA FOLLOWS
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TAXI AT IDLE THRUST

TIME (HRS)	RANGE (NM)	FUEL USED (LBS)	WEIGHT (LBS)	ALT. (FT)	FUEL FLOW (LB/HR)
0.0	0.	0.	2000.	0.	78.
0.183	0.	14.	1986.	0.	78.

*****START OF INPUT FOR MISSION DESCRIPTION
ISEG=2,DELTVR=2.5,HTMAX=400.,*

VSTLKT= 55.2 KTS EAS VRAT= 1.200 CLTO= 1.6661
VEND = 116.0 KNOTS EAS
518.6699 0.0

TAKEOFF (ELEVATION= 0. FT)

TIME (SEC)	DIST. (FEET)	FUEL USED (LBS)	WEIGHT (LBS)	ALT. (FT)	TAS (KTS)	EAS (KTS)	MACH NO.	ACCEL (FPS2)	CL	CD	ALPHA (DEG)	GAMMA (DEG)	ROC (FPM)	LD FA	THRUST (LBS)	FUEL FLOW (LB/HR)	FUS. ANGLE (DEG)
0.0	0.0	14.3	1985.	0.0	0.0	0.0	0.0	6.22	1.3049	0.0619	0.0	0.0	0.0	0.	424.	170.	0.0
1.0	3.1	14.3	1986.	0.0	3.7	3.7	0.006	6.14	1.3049	0.0619	0.0	0.0	0.0	0.	419.	170.	0.0
2.0	12.4	14.4	1986.	0.0	7.3	7.3	0.011	6.06	1.3049	0.0619	0.0	0.0	0.0	0.	414.	170.	0.0
3.0	27.7	14.4	1986.	0.0	10.9	10.9	0.016	5.97	1.3049	0.0619	0.0	0.0	0.0	0.	410.	170.	0.0
4.0	49.0	14.5	1986.	0.0	14.4	14.4	0.022	5.89	1.3049	0.0619	0.0	0.0	0.0	0.	405.	170.	0.0
5.0	75.2	14.5	1985.	0.0	17.9	17.9	0.027	5.80	1.3050	0.0619	0.0	0.0	0.0	0.	401.	170.	0.0
6.0	103.3	14.6	1985.	0.0	21.3	21.3	0.032	5.71	1.3050	0.0619	0.0	0.0	0.0	0.	397.	170.	0.0
7.0	148.0	14.6	1985.	0.0	24.6	24.6	0.037	5.62	1.3051	0.0619	0.0	0.0	0.0	0.	393.	170.	0.0
8.0	192.5	14.6	1985.	0.0	28.0	28.0	0.042	5.53	1.3051	0.0619	0.0	0.0	0.0	0.	389.	170.	0.0
9.0	242.9	14.7	1985.	0.0	31.2	31.2	0.047	5.44	1.3052	0.0619	0.0	0.0	0.0	0.	385.	171.	0.0
10.0	297.9	14.7	1985.	0.0	34.4	34.4	0.052	5.35	1.3052	0.0619	0.0	0.0	0.0	0.0	381.	171.	0.0
11.0	358.7	14.8	1985.	0.0	37.6	37.6	0.057	5.26	1.3053	0.0619	0.0	0.0	0.0	0.0	378.	171.	0.0
12.0	424.7	14.8	1985.	0.0	40.7	40.7	0.061	5.16	1.3054	0.0619	0.0	0.0	0.0	0.0	374.	171.	0.0
13.0	[illegible]	14.9	1985.	0.0	43.7	43.7	0.066	5.07	1.3055	0.0619	0.0	0.0	0.0	0.0	371.	171.	0.0
14.0	572.3	14.9	1985.	0.0	46.7	46.7	0.071	4.97	1.3056	0.0619	0.0	0.0	0.0	0.0	368.	171.	0.0
15.0	653.6	15.0	1985.	0.0	49.6	49.6	0.075	4.88	1.3056	0.0619	0.0	0.0	0.0	0.0	365.	171.	0.0
16.0	739.9	15.0	1985.	0.0	52.5	52.5	0.079	4.78	1.3057	0.0619	0.0	0.0	0.0	0.0	362.	171.	0.0
17.0	[illegible]	15.1	1985.	0.0	55.3	55.3	0.084	4.69	1.3058	0.0619	0.0	0.0	0.0	0.0	359.	171.	0.0
18.0	926.5	15.1	1985.	0.0	58.1	58.1	0.088	4.59	1.3059	0.0619	0.0	0.0	0.0	0.0	356.	171.	0.0
19.0	1027.0	15.2	1985.	0.0	60.8	60.8	0.092	4.50	1.3060	0.0619	0.0	0.0	0.0	0.0	353.	171.	0.0
20.0	1131.9	15.2	1985.	0.0	63.4	63.4	0.096	4.40	1.3061	0.0619	0.0	0.0	0.0	0.0	351.	171.	0.0
ROTATION (TIME= 20.7 AND TAS= 65.2 EAS= 65.2)																	
21.0	1241.2	15.3	1985.	0.0	66.0	66.0	0.100	4.31	1.3212	0.0624	0.17	0.0	0.0	0.78	348.	171.	0.17
22.0	[illegible]	15.3	1985.	0.0	68.5	68.5	0.103	4.16	1.4552	0.0578	1.83	0.0	0.0	0.93	345.	171.	1.83
LIFTOFF (TIME= 22.4 DIST= 1401.4 TAS= 69.5 EAS= 69.5)																	
23.0	1472.6	15.4	1985.	0.2	70.9	70.9	0.107	3.68	1.5956	0.0736	3.32	0.44	55.6	1.09	343.	171.	3.77
24.0	1593.9	15.4	1985.	3.3	72.6	72.7	0.110	1.95	1.6443	0.0864	3.69	2.81	360.2	1.19	341.	171.	6.50
25.0	1717.0	15.5	1985.	11.9	73.4	73.4	0.111	0.69	1.4790	0.0883	1.63	4.87	631.6	1.09	341.	171.	6.52
26.0	1841.6	15.5	1984.	23.6	73.6	73.6	0.111	0.16	1.3959	0.0875	0.67	5.83	737.3	1.04	340.	171.	6.50
DISTANCE TO 35 FT.= 1948.7																	
27.0	1964.3	15.5	1984.	36.7	73.6	73.6	0.111	0.00	1.3514	0.0852	0.19	6.20	806.4	1.01	340.	171.	6.39
28.0	2088.0	15.5	1984.	50.2	73.7	73.5	0.111	-0.00	1.3536	0.0854	0.21	6.20	806.0	1.01	340.	171.	6.41
29.0	2211.7	15.6	1984.	63.5	73.7	73.6	0.111	0.03	1.3484	0.0850	0.16	6.15	799.5	1.01	340.	171.	6.30
30.0	2335.4	15.7	1984.	76.9	73.7	73.6	0.111	0.00	1.3419	0.0845	0.09	6.22	808.7	1.00	340.	171.	6.30
31.0	2459.2	15.7	1984.	90.4	73.7	73.6	0.111	-0.00	1.3419	0.0845	0.09	6.23	810.2	1.00	340.	171.	[illegible]
32.0	2582.9	15.8	1984.	103.9	73.7	73.6	0.111	-0.00	1.3419	0.0845	0.09	6.23	810.0	1.00	340.	171.	[illegible]
33.0	2706.7	15.8	1984.	117.4	73.7	73.6	0.111	-0.01	1.3419	0.0845	0.09	6.23	810.3	1.00	339.	171.	6.32
34.0	2830.4	15.9	1984.	130.9	73.7	73.6	0.111	-0.01	1.3419	0.0845	0.09	6.22	809.9	1.00	339.	171.	6.31
35.0	2954.2	15.9	1984.	144.4	73.7	73.6	0.111	-0.00	1.3419	0.0845	0.09	6.22	808.9	1.00	339.	170.	6.30
36.0	[illegible]	16.0	1984.	157.9	73.7	73.6	0.111	0.00	1.3419	0.0845	0.09	6.20	807.3	1.00	339.	170.	6.29
37.0	3201.7	16.0	1984.	171.3	73.7	73.6	0.111	0.01	1.3419	0.0845	0.09	6.20	806.2	1.00	339.	170.	6.28
38.0	3325.5	16.1	1984.	184.8	73.7	73.6	0.111	0.01	1.3419	0.0845	0.09	6.19	805.9	1.00	339.	170.	6.28
39.0	3449.4	16.1	1984.	198.2	73.7	73.5	0.111	0.01	1.3419	0.0845	0.09	6.19	805.3	1.00	339.	170.	6.27
40.0	3573.2	16.2	1984.	211.6	73.7	73.5	0.111	0.01	1.3419	0.0845	0.09	6.19	805.4	1.00	339.	170.	6.27
41.0	3697.0	16.2	1984.	225.0	73.8	73.5	0.111	0.01	1.3419	0.0845	0.09	6.18	805.1	1.00	338.	170.	6.27
42.0	3820.9	16.3	1984.	238.4	73.8	73.5	0.111	0.01	1.3419	0.0845	0.09	6.18	804.5	1.00	338.	170.	6.27
43.0	3944.7	16.3	1984.	251.8	73.8	73.5	0.112	0.01	1.3419	0.0845	0.09	6.18	804.5	1.00	338.	170.	6.27
44.0	[illegible]	16.4	1984.	265.2	73.8	73.5	0.112	0.01	1.3419	0.0845	0.09	6.18	804.2	1.00	338.	170.	6.26
45.0	4192.5	16.4	1984.	278.6	73.8	73.5	0.112	0.01	1.3419	0.0845	0.09	6.17	803.6	1.00	338.	170.	6.26
46.0	4316.4	16.4	1984.	292.0	73.8	73.5	0.112	0.01	1.3419	0.0845	0.09	6.17	803.5	1.00	338.	170.	6.26
47.0	4440.3	16.5	1983.	305.4	73.8	73.5	0.112	0.01	1.3419	0.0845	0.09	6.16	803.0	1.00	338.	170.	6.25
48.0	4564.2	16.5	1983.	318.8	73.8	73.5	0.112	-0.01	1.3513	0.0852	0.19	6.15	802.2	1.00	337.	170.	6.35
49.0	4688.1	16.6	1983.	332.1	73.8	73.5	0.112	0.06	1.3398	0.0836	-0.03	6.10	794.9	0.99	337.	170.	6.07
50.0	4812.1	16.6	1983.	345.3	73.9	73.5	0.112	0.01	1.3639	0.0861	0.32	6.07	791.5	1.01	337.	170.	6.39
51.0	4936.1	16.7	1983.	358.4	73.9	73.5	0.112	0.06	1.3506	0.0851	0.18	6.03	786.4	1.00	337.	169.	6.21
[illegible]	[illegible]	16.7	1983.	[illegible]	73.9	73.5	0.112	[illegible]	1.3371	0.0841	[illegible]	6.02	785.6	0.99	337.	169.	6.06

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53.0	5184.4	16.8	1983.	384.8	73.9	73.6	0.112	-0.00	1.3511	0.0852	0.19	6.12	798.7	1.01	337.	169.	6.30
54.0	5308.5	16.8	1983.	398.1	74.0	73.6	0.112	0.01	1.3417	0.0845	0.09	6.12	799.0	1.00	337.	169.	6.21

ALL ENGINE DISTANCE TO 35 FT. (L) = 1948.7 FEET
FAR 25 T.O. DISTANCE (1.15*L) = 2241.0 FEET
ALL ENGINE DISTANCE TO 50 FT. = 2086.5 FEET

AT END OF TAKEOFF PHASE
TIME= 0.198 HRS FUEL USED= 17. LBS WEIGHT= 1983. LBS ALT.= 400. FT.

*****START OF INPUT FOR MISSION DESCRIPTION
ISEG*3,DELH*500.,*

CLIMB TO 10000. FT. AT MAXIMUM RATE OF CLIMB

TIME (HRS)	RANGE (NM)	FUEL USED (LBS)	WEIGHT (LBS)	ALT. (FT)	TAS (KTS)	EAS (KTS)	MACH NO.	MACH DIV	CL	CD	ALPHA (DEG)	GAMMA (DEG)	FUS. ANGLE (DEG)	R/C (FPM)	THRUST (LBS)	FUEL FLOW (LB/HR)
0.198	0.	17.	1983.	400.	117.	116.	0.176	0.621	0.5412	0.0325	1.85	4.65	6.50	959.	279.	157.
0.200	0.	17.	1983.	500.	117.	116.	0.177	0.621	0.5412	0.0325	1.85	4.65	6.50	960.	282.	159.
0.209	1.	18.	1981.	1000.	118.	116.	0.178	0.621	0.5408	0.0325	1.84	4.66	6.50	969.	281.	159.
0.217	2.	20.	1980.	1500.	119.	116.	0.180	0.621	0.5404	0.0325	1.84	4.65	6.50	977.	283.	161.
0.225	3.	21.	1979.	2000.	119.	115.	0.182	0.621	0.5400	0.0325	1.83	4.67	6.50	985.	283.	163.
0.234	4.	23.	1977.	2500.	120.	116.	0.183	0.620	0.5433	0.0326	1.87	4.63	6.50	982.	279.	161.
0.243	5.	24.	1976.	3000.	120.	115.	0.183	0.620	0.5502	0.0329	1.94	4.55	6.50	967.	275.	158.
0.251	6.	25.	1975.	3500.	120.	114.	0.184	0.619	0.5571	0.0332	2.01	4.49	6.50	953.	271.	156.
0.260	7.	27.	1973.	4000.	120.	113.	0.184	0.618	0.5660	0.0335	2.11	4.39	6.50	932.	267.	153.
0.269	8.	28.	1972.	4500.	120.	112.	0.184	0.617	0.5749	0.0339	2.20	4.30	6.50	912.	263.	151.
0.278	10.	29.	1971.	5000.	120.	112.	0.185	0.616	0.5819	0.0341	2.27	4.23	6.50	897.	259.	149.
0.288	11.	31.	1969.	5500.	120.	111.	0.185	0.615	0.5910	0.0345	2.37	4.13	6.50	876.	256.	147.
0.297	12.	32.	1968.	6000.	120.	110.	0.185	0.614	0.6001	0.0349	2.47	4.03	6.50	856.	252.	145.
0.307	13.	34.	1966.	6500.	120.	109.	0.185	0.613	0.6092	0.0353	2.56	3.94	6.50	835.	248.	143.
0.317	14.	35.	1965.	7000.	120.	108.	0.186	0.612	0.6184	0.0357	2.65	3.84	6.50	814.	244.	141.
0.327	15.	37.	1963.	7500.	120.	107.	0.186	0.611	0.6256	0.0360	2.74	3.76	6.50	800.	241.	138.
0.337	17.	38.	1962.	8000.	120.	107.	0.187	0.610	0.6349	0.0364	2.83	3.67	6.50	779.	237.	136.
0.348	18.	39.	1961.	8500.	120.	105.	0.187	0.609	0.6429	0.0368	2.92	3.58	6.50	762.	233.	134.
0.359	19.	41.	1959.	9000.	120.	105.	0.188	0.608	0.6530	0.0372	3.03	3.47	6.50	739.	230.	132.
0.370	21.	42.	1958.	9500.	120.	104.	0.188	0.607	0.6610	0.0376	3.11	3.39	6.50	722.	226.	130.
0.382	22.	44.	1956.	10000.	121.	104.	0.189	0.606	0.6692	0.0380	3.20	3.30	6.50	704.	223.	128.

END OF CLIMB TO 10000. FT
TIME= 0.382 HRS FUEL USED= 44. LBS WEIGHT= 1956. LBS RANGE= 22. NM

*****START OF INPUT FOR MISSION DESCRIPTION
ISEG*9,*

*****START OF INPUT FOR CONTROL
NPC*1,NSC*8,*

*****START OF INPUT FOR RESERVE OR RANGE INFO.
FRESF*1.0,
*

*****START OF INPUT FOR CONTROL
NPC*2,NSC*8,
EM*.2041,ALT*10000.,
*

SUMMARY OF CRUISE LIFT-WEIGHT BALANCE
ANGLE OF ATTACK(DEGREES) 2.192 LIFT 1956.1 L/D 16.991 ALTITUDE 10000.0

SUMMARY OF CRUISE LIFT-WEIGHT BALANCE
ANGLE OF ATTACK(DEGREES) 1.102 LIFT 1605.2 L/D 15.708 ALTITUDE 10000.0

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CRUISE SUMMARY

TIME (HRS)	RANGE (NM)	FUEL USED (LBS)	WEIGHT (LBS)	ALT. (FT)	TAS (KTS)	EAS (KTS)	CL	ANGLE ATTACK (DEG)	FUS. ANGLE (DEG)	L/D	MACH NO.	MACH DIV	FUEL FLOW (LB/HR)	BREG FACT (NM)	
0.382	22.	44.	1956.	10000.	130.	112.	0.5761	2.192	2.19	16.991	0.204	0.617	71.	3589.	BEGIN
5.517	692.	394.	1605.	10000.	130.	112.	0.4730	1.102	1.10	15.708	0.204	0.629	65.	3212.	END

RESERVE FUEL = 53. LBS.

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*****START OF INPUT FOR CONTROL

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*****START OF INPUT FOR RANGE REQUIREMENT
       RCRRG*800.*          FACA1*.9**

*****START OF INPUT FOR CONTROL
     NPC*2.*
**************************************************************************************************************
ITERATION TO BALANCE RANGE
 RANGE ERROR, RANGE ERROR MINUS 1 -0.1350  1.0000
 GROSS WGT, GROSS WGT MINUS1          2222.2      2000.0

TAKEOFF**  DELCL=  0.9753  DELCD=  0.0221  CLMAX=  2.4055  DELTA6=  0.9900  SIGMTO =  0.285

LANDING**  DELCL=  2.3534  DELCD=  0.1492  CLMAX=  3.7836  DELTA6=  0.8550  SIGMLD =  0.580

SUMMARY OF CRUISE LIFT-WEIGHT BALANCE
  ANGLE OF ATTACK(DEGREES)      2.329 LIFT     2222.2 L/D     17.998 ALTITUDE    10000.0

VSTLKT=  55.3 KTS EAS    VRAT= 1.200 CLTO= 1.6755
     VEND =  119.6 KNOTS EAS

ROTATION (TIME=    21.1 AND TAS=      65.3 EAS=      65.3)
LIFTOFF (TIME=    22.8 DIST=    1423.4 TAS=      69.4 EAS=      69.5)
DISTANCE TO 35 FT.=  1977.6

ALL ENGINE DISTANCE TO 35 FT. (L) =        1977.6 FEET
FAR 25 T.O. DISTANCE (1.15*L) =       2274.3 FEET
ALL ENGINE DISTANCE TO 50 FT. =       2117.6 FEET

   AT END OF TAKEOFF PHASE
TIME=  0.015 HRS    FUEL USED=          3. LBS    WEIGHT=     2219. LBS    ALT.=       400. FT.

RESIZE ENGINES AT CRUISE TO ACCOUNT FOR RESIZED NACELLES

PROPULSION SYSTEM WEIGHTS
  ENGINE WEIGHT/ENGINE                        93.5
  NACELLE WEIGHT/ENGINE                       26.7
  PYLON WEIGHT/ENGINE                          0.0
     PROP OR OFAN                 0.0
    GEARBOX                       0.0
    SHROUD                        0.0

ENGINE POD DIMENSIONS
  ENGINE FACE DIAMETER(FT)                   1.49
  NACELLE LENGTH(FT)                         5.27


VSTLKT=  55.3 KTS EAS    VRAT= 1.200 CLTO= 1.6755
     VEND =  119.5 KNOTS EAS

ROTATION (TIME=    21.1 AND TAS=      65.3 EAS=      65.3)
LIFTOFF (TIME=    22.8 DIST=    1423.7 TAS=      69.5 EAS=      69.5)
DISTANCE TO 35 FT.=  1977.6

ALL ENGINE DISTANCE TO 35 FT. (L) =        1977.6 FEET
FAR 25 T.O. DISTANCE (1.15*L) =       2274.3 FEET
ALL ENGINE DISTANCE TO 50 FT. =       2117.7 FEET

   AT END OF TAKEOFF PHASE
TIME=  0.015 HRS    FUEL USED=          3. LBS    WEIGHT=     2219. LBS    ALT.=       400. FT.
ENGINE SIZED TO MATCH CRUISE DRAG - SLS AIRFLOW=       26.69

ENGINE SIZED TO MATCH TAKE OFF DISTANCE OF 2000.0FEET - SLS AIRFLOW=       29.83

RATED SEA LEVEL STATIC THRUST PER ENGINE=      467.6 LBS
**************************************************************************************************************

PROPULSION SYSTEM WEIGHTS
  ENGINE WEIGHT/ENGINE                        93.5
  NACELLE WEIGHT/ENGINE                       26.7
  PYLON WEIGHT/ENGINE                          0.0
     PROP OR OFAN                 0.0
    GEARBOX                       0.0
    SHROUD                        0.0

ENGINE POD DIMENSIONS
  ENGINE FACE DIAMETER(FT)                   1.49
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NACELLE LENGTH(FT)                    5.27

 WING LOCATION INFO.                         H-TAIL VOL. ARM         =     9.31        C.G.LOCATION OF PROPULSION=    14.42
FUSELAGE LENGTH          =    21.91          H-TAIL C.G.LOCATION     =    18.81        C.G.OF REMAINING WEIGHT   =     7.89
WING 1/4C LOC.ON C.L.=         9.07          H-TAIL MAC FROM C.L.    =     2.73
MAC 1/4C LOCATION        =     9.13          H-TAIL LOCAT ON VERT.=        0.0
MAC DIST.FROM C.L.       =     8.16          V-TAIL VOL. ARM         =     9.97
WING C.G.LOCATION        =     9.40          V-TAIL C.G.LOCATION     =    19.48
TIP TANKS C.G.LOCATE =         0.0

 AIRCRAFT C.G. LOCATION =     9.13 FT. OR 0.250 OF MAC
                        WING          H-TAIL          V-TAIL

 AREA                  88.889         28.845          22.701
 SPAN                  32.660         12.318           4.997
 ASPECT RATIO          12.000          5.260           1.100
 TAPER RATIO            1.000          0.500           0.500
 1/4C. SWEEP            0.0           27.000          32.000
 L.E. SWEEP             0.0           29.808          42.858
 C.L. CHORD             2.722          3.122           6.057
 MEAN CHORD             2.722          2.428           4.711
 TIP CHORD              2.722          1.561           3.029

   TAXI AT IDLE THRUST
                       FUEL                            FUEL
 TIME     RANGE        USED      WEIGHT     ALT.       FLOW
 (HRS)    (NM)         (LBS)     (LBS)      (FT)      (LB/HR)

 0.0         0.          0.       2222.       0.        85.
 0.193       0.         16.       2207.       0.        85.

 VSTLKT=  55.1 KTS EAS    VRAT= 1.200 CLTO= 1.6755
     VEND =  119.4 KNOTS EAS

 ROTATION (TIME=    20.9 AND TAS=       65.1 EAS=      65.1)
 LIFTOFF (TIME=    22.4 DIST=    1386.2 TAS=      68.9 EAS=      68.9)
 DISTANCE TO 35 FT.=  1959.9

 ALL ENGINE DISTANCE TO 35 FT. (L) =      1959.9 FEET
 FAR 25 T.O. DISTANCE (1.15*L) =       2253.9 FEET
 ALL ENGINE DISTANCE TO 50 FT. =       2099.0 FEET

    AT END OF TAKEOFF PHASE
 TIME=  0.199 HRS    FUEL USED=         19. LBS     WEIGHT=      2204. LBS     ALT.=        400. FT.

    END OF CLIMB TO     10000. FT
 TIME=  0.379 HRS    FUEL USED=         48. LBS     WEIGHT=      2174. LBS     RANGE=        22. NM

    *************************************************************************************************************

    CRUISE SUMMARY
                                                                      ANGLE     FUS.                MACH     MACH      FUEL       BREG
                                                                      ATTACK    ANGLE      L/D      NO.      DIV       FLOW       FACT
                        FUEL                                  CL      (DEG)     (DEG)                                  (LB/HR)    (NM)
 TIME      RANGE        USED     WEIGHT     ALT.      TAS     EAS                                                                          
 (HRS)     (NM)         (LBS)    (LBS)      (FT)      (KTS)   (KTS)                                                                        
 0.379       22.         48.     2174.     10000.     130.    112.  0.5763   2.194    2.19     17.511   0.204    0.617      77.       3681.    BEGIN
 7.001     [illegible]  531.     1691.     10000.     130.    112.  0.4481   0.839    0.84     15.832   0.204    0.531      69.       3191.    END

       RESERVE FUEL =        58. LBS.

                                                                                *************************************************
    *****************************************************************************************************************
 *********************************************************************************************************************
 ITERATION TO BALANCE RANGE
 RANGE ERROR, RANGE ERROR MINUS 1    0.1073  -0.1350
 GROSS WGT, GROSS WGT MINUS1          2123.8        2222.2

 TAKEOFF**   DELCL=   0.9720   DELCD=   0.0220   CLMAX=   2.3996   DELTA6=   0.9900   SIGMTO =   0.285

 LANDING**   DELCL=   2.3454   DELCD=   0.1489   CLMAX=   3.7730   DELTA6=   0.8550   SIGMLD =   0.589

 SUMMARY OF CRUISE LIFT-WEIGHT BALANCE
   ANGLE OF ATTACK(DEGREES)       2.329 LIFT     2123.8 L/D     17.712 ALTITUDE    10000.0

 VSTLKT=   55.4 KTS EAS    VRAT= 1.200 CLTO= 1.6714
     VEND =   119.7 KNOTS EAS
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ROTATION (TIME=   21.4 AND TAS=     65.3 EAS=     65.4)
LIFTOFF (TIME=   23.0 DIST=   1432.4 TAS=     69.2 EAS=     69.3)
DISTANCE TO 35 FT.=  2003.5

 ITERATION TO MATCH TAKEOFF DISTANCE
  ATO,ATORG,,ASLS  2003.  2000.   28.23

PROPULSION SYSTEM WEIGHTS
  ENGINE WEIGHT/ENGINE                        88.5
  NACELLE WEIGHT/ENGINE                       25.3
  PYLON WEIGHT/ENGINE                          0.0
      PROP OR OFAN                 0.0
    GEARBOX                        0.0
    SHROUD                         0.0

ENGINE POD DIMENSIONS
  ENGINE FACE DIAMETER(FT)                  1.45
  NACELLE LENGTH(FT)                        5.13

ENGINE SIZED TO MATCH CRUISE DRAG - SLS AIRFLOW=       25.69

ENGINE SIZED TO MATCH TAKE OFF DISTANCE OF 2000.0FEET - SLS AIRFLOW=      28.23

RATED SEA LEVEL STATIC THRUST PER ENGINE=    442.5 LBS
*****************************************************************************************************************

PROPULSION SYSTEM WEIGHTS
  ENGINE WEIGHT/ENGINE                        88.5
  NACELLE WEIGHT/ENGINE                       25.3
  PYLON WEIGHT/ENGINE                          0.0
      PROP OR OFAN                 0.0
    GEARBOX                        0.0
    SHROUD                         0.0

ENGINE POD DIMENSIONS
  ENGINE FACE DIAMETER(FT)                  1.45
  NACELLE LENGTH(FT)                        5.13

 WING LOCATION INFO.
FUSELAGE LENGTH        =    21.91         H-TAIL VOL. ARM         =      9.52         C.G.LOCATION OF PROPULSION=     14.42
WING 1/4C LOC.ON C.L.=       9.02         H-TAIL C.G.LOCATION     =     18.94         C.G.OF REMAINING WEIGHT    =     7.89
MAC 1/4C LOCATION      =     9.06         H-TAIL MAC FROM C.L.    =      2.61
MAC DIST.FROM C.L.     =     7.98         H-TAIL LOCAT ON VERT.=         0.0
WING C.G.LOCATION      =     9.33         V-TAIL VOL. ARM         =     10.16
TIP TANKS C.G.LOCATE =       0.0          V-TAIL C.G.LOCATION     =     19.58

 AIRCRAFT C.G. LOCATION =      9.06 FT. OR 0.250 OF MAC

                          WING          H-TAIL          V-TAIL

  AREA                   84.952         26.362          20.829
  SPAN                   31.928         11.776           4.787
  ASPECT RATIO           12.000          5.260           1.100
  TAPER RATIO             1.000          0.500           0.500
  1/4C. SWEEP             0.0           27.000          32.000
  L.E. SWEEP              0.0           29.808          42.858
  C.L. CHORD              2.661          2.985           5.802
  MEAN CHORD              2.661          2.322           4.513
  TIP CHORD               2.661          1.492           2.901

   TAXI AT IDLE THRUST

                       FUEL                              FUEL
  TIME     RANGE       USED      WEIGHT      ALT.        FLOW
  (HRS)    (NM)        (LBS)     (LBS)       (FT)      (LB/HR)

  0.0          0.          0.     2124.          0.        81.
  0.183        0.         15.     2109.          0.        81.

VSTLKT=  55.2 KTS EAS    VRAT= 1.200 CLTO= 1.6714
      VEND =  115.9 KNOTS EAS

ROTATION (TIME=   21.2 AND TAS=     65.1 EAS=     65.2)
LIFTOFF (TIME=   22.8 DIST=   1418.1 TAS=     69.1 EAS=     69.1)
DISTANCE TO 35 FT.=  1981.3
```

```
ALL ENGINE DISTANCE TO 35 FT. (L) =      1981.3 FEET
FAR 25 T.O. DISTANCE (1.15*L) =       2278.4 FEET
ALL ENGINE DISTANCE TO 50 FT. =       2122.9 FEET

  AT END OF TAKEOFF PHASE
TIME=  0.199 HRS   FUEL USED=        18. LBS   WEIGHT=    2106. LBS   ALT.=     400. FT.

  END OF CLIMB TO   10000. FT
TIME=  0.366 HRS   FUEL USED=        46. LBS   WEIGHT=    2077. LBS   RANGE=     22. NM

*****************************************************************************************************

  CRUISE SUMMARY
```

TIME (HRS)	RANGE (NM)	FUEL USED (LBS)	WEIGHT (LBS)	ALT. (FT)	TAS (KTS)	EAS (KTS)	CL	ANGLE ATTACK (DEG)	FUS. ANGLE (DEG)	L/D	MACH NO.	MACH DIV	FUEL FLOW (LB/HR)	BREG FACT (NM)	
0.366	22.	46.	2077.	10000.	130.	112.	0.5761	2.193	2.19	17.295	0.204	0.617	74.	3657.	BEGIN
6.427	311.	472.	1651.	10000.	130.	112.	0.4580	0.943	0.94	15.778	0.204	0.630	67.	3212.	END

```
    RESERVE FUEL =       56. LBS.

  ***************************************************************************************************
***************************************************************************************************************
ITERATION TO BALANCE RANGE
 RANGE ERROR, RANGE ERROR MINUS 1   0.0132   0.1073
 GROSS WGT, GROSS WGT MINUS1        2110.0      2123.8

TAKEOFF**  DELCL=  0.9715  DELCD=  0.0220  CLMAX=  2.3988  DELTAG=  0.9900  SIGMTO =  0.285

LANDING**  DELCL=  2.3442  DELCD=  0.1488  CLMAX=  3.7715  DELTAG=  0.8550  SIGMLD =  0.580

SUMMARY OF CRUISE LIFT-WEIGHT BALANCE
  ANGLE OF ATTACK(DEGREES)      2.329 LIFT     2110.0 L/D     17.671 ALTITUDE    10000.0

VSTLKT=  55.4 KTS EAS    VRAT= 1.200 CLTO= 1.6708
     VEND =  119.7 KNOTS EAS

ROTATION (TIME=   21.6 AND TAS=     65.3 EAS=     65.4)
LIFTOFF (TIME=    23.2 DIST=    1442.8 TAS=     69.1 EAS=     69.2)
DISTANCE TO 35 FT.=  2022.2

  ITERATION TO MATCH TAKEOFF DISTANCE
   XTO,XTORQ,WASLS  2022.  2000.   27.80

VSTLKT=  55.4 KTS EAS    VRAT= 1.200 CLTO= 1.6708
     VEND =  120.3 KNOTS EAS

ROTATION (TIME=   19.3 AND TAS=     65.3 EAS=     65.4)
LIFTOFF (TIME=    21.0 DIST=    1316.7 TAS=     69.9 EAS=     69.9)
DISTANCE TO 35 FT.=  1854.4

  ITERATION TO MATCH TAKEOFF DISTANCE
   XTO,XTORQ,WASLS  1854.  2000.   30.58

VSTLKT=  55.4 KTS EAS    VRAT= 1.200 CLTO= 1.6708
     VEND =  119.7 KNOTS EAS

ROTATION (TIME=   21.3 AND TAS=     65.3 EAS=     65.4)
LIFTOFF (TIME=    23.0 DIST=    1436.5 TAS=     69.5 EAS=     69.5)
DISTANCE TO 35 FT.=  1994.5

  ITERATION TO MATCH TAKEOFF DISTANCE
   XTO,XTORQ,WASLS  1995.  2000.   28.17

PROPULSION SYSTEM WEIGHTS
  ENGINE WEIGHT/ENGINE                    88.3
  NACELLE WEIGHT/ENGINE                   25.2
  PYLON WEIGHT/ENGINE                      0.0
      PROP OR GEAN              0.0
     GEARBOX                    0.0
     SHROUD                     0.0
```


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ENGINE POD DIMENSIONS
  ENGINE FACE DIAMETER(FT)             1.45
  NACELLE LENGTH(FT)                   5.12

ENGINE SIZED TO MATCH CRUISE DRAG - SLS AIRFLOW=      25.56
ENGINE SIZED TO MATCH TAKE OFF DISTANCE OF 2000.0FEET - SLS AIRFLOW=      28.17
RATED SEA LEVEL STATIC THRUST PER ENGINE=    441.5 LBS
*******************************************************************************************************

PROPULSION SYSTEM WEIGHTS
  ENGINE WEIGHT/ENGINE                   88.3
  NACELLE WEIGHT/ENGINE                  25.2
  PYLON WEIGHT/ENGINE                     0.0
      PROP OR DEAN             0.0
      GEARBOX                 0.0
      SHROUD                  0.0

ENGINE POD DIMENSIONS
  ENGINE FACE DIAMETER(FT)             1.45
  NACELLE LENGTH(FT)                   5.12

 WING LOCATION INFO.
FUSELAGE LENGTH       =   21.91        H-TAIL VOL. ARM          =     9.54        C.G.LOCATION OF PROPULSION=   14.42
WING 1/4C LOC.ON C.L.=     9.01        H-TAIL C.G.LOCATION      =    18.95        C.G.OF REMAINING WEIGHT    =    7.89
MAC 1/4C LOCATION     =    9.05        H-TAIL MAC FROM C.L.     =     2.59
MAC DIST.FROM C.L.    =    7.96        H-TAIL LOCAT ON VERT.=         0.0
WING C.G.LOCATION     =    9.32        V-TAIL VOL. ARM          =    10.18
TIP TANKS C.G.LOCATE  =    0.0         V-TAIL C.G.LOCATION      =    19.59

 AIRCRAFT C.G. LOCATION =      9.05 FT. OR 0.250 OF MAC

                       WING        H-TAIL        V-TAIL

  AREA                84.401       26.036        20.581
  SPAN                31.825       11.702         4.758
  ASPECT RATIO        12.000        5.260         1.100
  TAPER RATIO          1.000        0.500         0.500
  1/4C. SWEEP          0.0         27.000        32.000
  L.E. SWEEP           0.0         29.868        42.858
  C.L. CHORD           2.652        2.966         5.767
  MEAN CHORD           2.652        2.307         4.486
  TIP CHORD            2.652        1.483         2.884

    TAXI AT IDLE THRUST

                    FUEL                        FUEL
 TIME     RANGE     USED     WEIGHT    ALT.     FLOW
 (HRS)    (NM)      (LBS)    (LBS)     (FT)    (LB/HR)

 0.0        0.        0.     2110.       0.       81.
 0.183      0.       15.     2095.       0.       81.

VSTLKT=  55.2 KTS EAS   VRAT= 1.200 CLTO= 1.6708
     VELD =  116.0 KNOTS EAS

ROTATION (TIME=    21.1 AND TAS=     65.1 EAS=     65.2)
LIFTOFF (TIME=   22.6 DIST=   1900.8 TAS=     68.9 EAS=     68.9)
DISTANCE TO 35 FT.=  1976.9

ALL ENGINE DISTANCE TO 35 FT. (L) =      1976.9 FEET
FAR 25 T.O. DISTANCE (1.15*L) =      2273.4 FEET
ALL ENGINE DISTANCE TO 50 FT. =      2118.9 FEET

   AT END OF TAKEOFF PHASE
TIME=  0.199 HRS    FUEL USED=        18. LBS    WEIGHT=      2092. LBS    ALT.=      400. FT.

   END OF CLIMB TO    10000. FT
TIME=  0.365 HRS    FUEL USED=        46. LBS    WEIGHT=      2064. LBS    RANGE=       22. NM

   *****************************************************************************************************

   CRUISE SUMMARY

                     FUEL                                               ANGLE    FUS.             MACH    MACH     FUEL     BREG
 TIME     RANGE      USED     WEIGHT    ALT.    TAS     EAS     CL     ATTACK   ANGLE    L/D      NO.     DIV     FLOW     FACT
 (HRS)    (NM)       (LBS)    (LBS)     (FT)   (KTS)   (KTS)           (DEG)    (DEG)                             (LB/HR)   (NM)
```

0.385	22.	46.	2064.	10000.	130.	112.	0.5761	2.193	2.19	17.262	0.204	0.617	74.	3648.	BEGIN
6.319	797.	463.	1647.	10000.	130.	112.	0.4597	0.961	0.96	15.772	0.204	0.630	67.	3211.	END

RESERVE FUEL = 55. LBS.

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GROSS WEIGHT = 2110. PASSENGERS = 2. PLUS CREW OF 1

FUSELAGE	LENGTH	(FLF)	21.91	FT
	WIDTH	(SWF)	4.33	FT
	WETTED AREA	(SF)	223.	SQFT
	DELTA P	(DELP)	0.0	PSI
WING	ASPECT RATIO	(AR)	12.00	
	AREA	(SW)	84.4	SQFT
	SPAN	(B)	31.8	FT
	GEOM. MEAN CHORD	(CBARW)	2.65	FT
	QUARTER CHORD SWEEP	(DLMC4)	0.0	DEG
	TAPER RATIO	(SLM)	1.000	
	ROOT THICKNESS	(TCR)	0.170	
	TIP THICKNESS	(TCT)	0.170	
	WING LOADING	(WGS)	25.0	PSF
	WING FUEL VOLUME	(VEA)	24.1	CUFT
HOR. TAIL	ASPECT RATIO	(ARHT)	5.26	
	AREA	(SHT)	26.0	SQFT
	SPAN	(BHT)	11.70	FT
	MEAN CHORD	(CBARHT)	2.31	FT
	THICKNESS/CHORD	(TCHT)	0.100	
	MOMENT ARM	(ELTH)	9.5	FT
	VOLUME COEFF.	(VBARH)	1.110	
VERT. TAIL	ASPECT RATIO	(ARVT)	1.10	
	AREA	(SVT)	20.6	SQFT
	SPAN	(BVT)	4.76	FT
	MEAN CHORD	(CBARVT)	4.49	FT
	THICKNESS/CHORD	(TCVT)	0.100	
	MOMENT ARM	(ELTV)	10.2	FT
	VOLUME COEFF.	(VBARV)	0.078	
ENG.NACELLES	LENGTH	(FLN)	5.12	FT
	MEAN DIAMETER	(DBARN)	1.45	FT
	NUMBER ENGINES	(ELP)	1.0	
	WETTED AREA	(SN)	23.34	SQFT

```
VDIVE =  241. KTS   VMO =  205. KTS   MMO = 0.616
ULT. LF =  5.70   MAN. LF =  3.80   GUST LF =  3.69

PROPULSION GROUP
    PRIMARY ENGINES              (WEP)        88.
    PRIMARY ENGINE INSTL.        (WPEI)       12.
    FUEL SYSTEM                  (WFSS)       31.
    PROPULSOR WEIGHT             (WPROP)       0.
    TOTAL PROP.GROUP WT.         (WP)        131.

STRUCTURES GROUP
    WING                         (WW)        234.
    HOR. TAIL                    (WHT)        47.
    VERT. TAIL                   (WVT)         0.
    FUSELAGE                     (WB)        182.
    LANDING GEAR                 (WLG)       114.
    PRIMARY ENG. SECTION         (WPES)       25.
    GROUP WEIGHT INC.            (DELWST)      0.
    TOTAL STRUC.GROUP WT.        (WST)       602.

FLIGHT CONTROLS GROP
    COCPIT CONTROLS              (WCC)        15.
    FIXED WING CONTROLS          (WCFW)       29.
    SAS                          (WSAS)        0.
    GROUP WEIGHT INC.            (DELWFC)      0.
    TOTAL CONTROL WT.            (WFC)        44.

WT. OF FIXED EQUIPMENT           (WFE)       214.

WEIGHT EMPTY                     (WE)        991.

FIXED USEFUL LOAD                (WFUL)      200. (INC. CREW OF 1)

OPERATING WEIGHT EMPTY           (OWE)      11[illegible]1.

PAYLOAD                          (WPL)      [illegible]0. (PAX=  2.)

FUEL                             (WFA)       519. (WFW=  519.) (WFTP=     0.)

GROSS WEIGHT                     (WG)       2110.
```

CRUISE MACH = 0.204   CRUISE ALTITUDE = 10000.

CRUISE RE.NUM. PER FT.   = 1.094E 06   FLATPLATE CF AT RE=10EX7 IS 0.00291

AERODYNAMIC DATA

DRAG BREAKDOWN	FLATPLATE AREA(SQFT)	CDO	WETTED AREA(SQFT)
WING	0.6044	0.00716	157.31
FUSELAGE	0.7413	0.00878	222.65
VERT.TAIL	0.0	0.0	41.16
HOR. TAIL	0.2551	0.00302	52.07
ENGINE NAC.	0.0247	0.00029	23.34
TIP TANKS	0.0	0.0	0.0
INCREMENTAL	0.0	0.0	0.0
FIXED GEAR	0.2087	0.00247	NOT INCL.
TOTAL	1.8342	0.02173	496.54

MEAN SKIN FRICTION COEF.= 0.003694

AERODYNAMIC COEFF.

A1		0.6832
A2		-0.1155
A3		0.0783
A4=.75X(T/C)		0.1275
A5=CDO--		0.0143
A6		2.4627
A7=1/(PI.SEE.AR)		0.0351
3-D LIFT SLOPE AT CRUISE MACH	(CLALPH)	5.4179 PER RADIAN
OSWALD FACTOR	(SEE)	0.7562

CRUISE CD = 0.0217 + 0.0351 CL**2

LOW SPEED LIFT/DRAG-GR.UP(IF RID.G.E.

	FLAPS UP			TAKEOFF			LANDING		
ALPHA	CL	CD	L/D	CL	CD	L/D	CL	CD	L/D
-2.00000	0.17819	0.02285	7.79967	1.15056	0.07077	16.25787	2.52331	0.22610	11.16019
0.0	0.36576	0.02642	13.84148	1.33907	0.08370	15.99846	2.71182	0.24556	11.04356
2.00000	0.55333	0.03247	17.04010	1.52758	0.09915	15.40696	2.90032	0.26793	10.82495
4.00000	0.74090	0.04099	18.07611	1.71608	0.11712	14.65296	3.08863	0.29322	10.53424
6.00000	0.92847	0.05197	17.86494	1.90459	0.13760	13.84149	3.27734	0.32142	10.19634
8.00000	1.11604	0.06542	17.05865	2.09310	0.16060	13.03272	3.46584	0.35254	9.83098
10.00000	1.30361	0.08134	16.02588	2.28160	0.18612	12.25848	3.65435	0.38658	9.45304

*****START OF INPUT FOR CONTROL
NPC=1,NSC=9,*

*****START OF INPUT FOR COSTING INFORMATION
UCSP=6=25.,THD=3000.,SRPM=0.0,
*

*****START OF INPUT FOR CONTROL
NPC=2,NSC=9,*

```
                    *** COST DATA ***

          ENGINES        NUMBER =        1.  TYPE=      10

          EMPTY WEIGHT=       991. LBS                   MAX.CRUISE SPEED=    130. KNOTS

          CONSUMER PRICE=  27918. DOL.          BASIC PRICE=  27918. DOL.
                                                ADD. EQUIPMENT COST=         0. DOL.

          DIRECT LABOR (   380.MHRS.)         1293.
          LABOR OVERHEAD(132.PCT)             1707.
          AIRFRAME MATERIALS                   550.
          PURCHASED EQUIP.                   11556.
              (ENGINE= 10974.)
              (PROP. =      0.)
              (OTHER =    582.)
                                             15107.   SUB-TOTAL
          ENG.TL.SALES.G-A( 31.PCT)           4612.
                                             19718.   MANUFACTURING COST
          FACTORY PROFIT(  9.PCT)             1757.
                                             21475.   DEALER COST
          DEALER-DIST. MARKUP( 30.PCT)        6443.
                                             27918.   BASIC PRICE

********************************************************************************************************

          RANGE=      797. N.M.     BLOCK FUEL=      463. LBS     BLOCK TIME=  6.319 HRS.

          FUEL RATE= 10.9 GPH.     TBO= 3000. HRS.     HOURS/INSP.=  100. HRS.

          VARIABLE COST      (DOL/HR)                 FIXED COST        (DOL/YR)
               FUEL+OIL           5.71                    STORAGE             0.
               INSP.+MAIN.       15.00                    INSURANCE         773. (HULL 2.0PCT)
               OVERHAUL RES.      0.80                    DEPRECIATION     2792. ( 8.YR-20.PCT)
               OTHER              0.0                     OTHER               0.
                                                          CREW                0.(OVERHEAD 50.PCT)
                                                          FAA TAX            25.
                             21.52 TOTAL                                   3590. TOTAL

          UTILIZATION(HRS/YR)           100.       200.       300.       400.       500.       800.
          TOTAL OPR.COST(DOL/HR)       57.42      39.47      33.49      30.49      28.70      26.01

*****START OF INPUT FOR CONTROL

END OF INPUT DATA. JOB COMPLETE.
```

APPENDIX B - THE WGHT MODULE IN GASP

A study was made of the weight estimation methodology embodied in the WGHT subroutine in GASP. This was concentrated on the wing weight calculation, because older methods are available which can be compared with it.

The WGHT method has advantages over older methods; it provides for weight reduction from the bending relief due to wing mounted masses, and it accounts for different types of high lift devices. However, it does not account for the effects of sweep. The major problem with it is that the results are consistently too high for values of the input parameters typical of light aircraft. Presumably this is because it was developed from a statistical population composed primarily of fighters, bombers, and transports. Past users have circumvented this problem by inputting a smaller value for SKWW, the trend equation constant. This requires some degree of foreknowledge, however, of what the correct answer should be. The default value should be one that will produce reasonable answers when used without modification by a naive programmer.

A brief study was run to compare this methodology with an older system, using identical data. The older system was developed at Beech, using fighters and transports in addition to the Beech data. It has since then been used to correctly calculate wing weight of several Cessna airplanes as well as the Learjet Model 35. It does not account for bending relief due to wing mounted masses, nor for different types of flaps, but it does handle sweep. Neither system has been checked at extremely high aspect ratios (~20).

The results calculated by the older method compared to those by the GASP method produced weight ratios ranging from .758 to .560 for the default value of SKWW = 220. The principal factor in this variation seemed to be aspect ratio, which was varied from 6 to 20. This indicates that the two systems vary excessively in their handling of this parameter. There are no data, how-

ever, to indicate which is correct. A second run was made with SKWW reduced to 132 and a simplified calculation of the initial wing weight. This produced comparison ratios ranging from 1.121 to 0.820. Aspect ratio again appeared to be the primary factor. A more extensive study is beyond the scope of this review, and would be of little use without more data to back it up.

The subroutine can be simplified in several minor ways. Several statements and variables can be eliminated by simply setting WW1 equal to 15% of the gross weight. At present it is found by a complex calculation procedure. This is unwarranted, since it is only used for the initial value in an iterative calculation.

APPENDIX C - COST ANALYSIS METHODOLOGY

A method commonly used by the General Aviation Industry for estimating new design aircraft costs for development and production employs AMPR (Airframe Manufacturer's Production Responsibility) weight to determine a parameter to which labor manhours and material costs can be associated through historical experience. AMPR weight may include only the airframe or it may be an aggregate weight of airframe and certain systems; however, engines, propellers, avionics and add-on equipment are not included. Thus, an aircraft's development and production cost can historically be accounted for in terms of manhours per unit weight of aircraft engineered, tested, tooled, and manufactured. The cost of manhours and materials is then easily scaled in terms of time and place according to any given manufacturer's experience and capabilities.

The cost analysis methodology for analysis and evaluation of candidate preliminary designs using the AMPR weight method is explained in the following paragraphs for estimating development, production unit costs, and operating costs.

Development and Certification Cost

Development and certification cost may be broken down into the following major cost items for purposes of estimation.

a. Engineering including burdened labor, special materials, purchased services, and flight testing.
b. Tooling including direct labor, overhead, materials (prototype soft tooling and all production tooling).
c. Manufacturing including direct labor, overhead, and materials.
d. Quality Assurance including direct labor and overhead.

Engineering Cost

Engineering development cost as function of AMPR weight may be formulated as follows:

$$C_{EL} = W_{AMPR} \times H_{ED} \times c_E \times K_{ESCE} \times K_{DF} \tag{1}$$

Where:

C_{EL}	=	cost of engineering labor to develop, test and certify the aircraft
W_{AMPR}	=	total AMPR weight in pounds
H_{ED}	=	engineering manhours per pound of AMPR weight
c_E	=	burdened cost of engineering labor
K_{ESCE}	=	labor cost escalation factor for engineering
K_{DF}	=	difficulty factor which is sometimes applied for increased complexity or difficulty in engineering and certification

$$C_{m/s} = \Sigma C_n \tag{2}$$

Where:

$C_{m/s}$	=	cost of materials and services
C_n	=	cost of item n material or purchased service such as wind tunnel models, wind tunnel tests, outside flight tests, flight test instrumentation, etc.

$$C_{FT} = H_{FT} \times c_{FT} \tag{3}$$

Where:

C_{FT}	=	cost of flight testing
H_{FT}	=	flight test hours for engineering development and certification flight testing
c_{FT}	=	cost per flight test hour

The total of engineering costs in a development certification program is the sum of all the above items.

$$C_E = C_{EL} + C_{m/s} = C_{FT} \tag{4}$$

Tooling Cost

Revised accounting methods now in use include sustaining tooling for on-going production as a part of manufacturing overhead. For a new model aircraft program, however, a separate estimate of tooling cost is needed to determine the burden for new tooling design and construction. Therefore, an accounting is made for tooling direct labor and overhead separate from that of manufacturing, and the manufacturing overhead rate is appropriately reduced. The

methodology for estimating tooling cost in a development program is described in the following.

Tooling cost as a function of AMPR weight may be formulated as follows:

$$C_T = C_{T_{DL}} + C_{T_{OH}} + C_{T_M} \tag{5}$$

Where:

$C_{T_{DL}}$ = cost of tooling direct labor

$C_{T_{OH}}$ = cost of tooling overhead

C_{T_M} = cost of tooling materials

$$C_{T_{DL}} = W_{AMPR} \times H_T \times c_T \times K_{ESC_{TL}} \tag{6}$$

Where:

W_{AMPR} = total AMPR weight in pounds

H_T = tooling manhours per pound of AMPR weight

c_T = cost per tooling direct labor manhour

$K_{ESC_{TL}}$ = cost escalation factor for tooling labor

$$C_{T_{OH}} = C_{T_{DL}} \times \frac{O_T}{100} \tag{7}$$

Where:

O_T = tooling overhead factor

$$C_{T_M} = W_{AMPR} \times H_T \times c_{T_M} \times K_{ESC_{TM}} \tag{8}$$

Where:

H_T = tooling manhours per pound of AMPR weight, as explained above

c_{T_M} = cost per pound of tooling materials

$K_{ESC_{TM}}$ = escalation factor for tooling materials

Manufacturing Cost

Manufacturing costs for prototype flight and static test articles may be formulated as follows:

$$C_M = C_{MDL} + C_{MOH} + C_{MM} \tag{9}$$

Where:

C_{MDL}	=	cost of manufacturing direct labor
C_{MOH}	=	cost of manufacturing overhead
C_{MM}	=	cost of manufacturing materials

$$C_{MDL} = np \times W'AMPR \times H_M \times c_M \times K_{ESCM} + nta \times W''AMPR \times H_M \times c_M \times K_{ESCM} \tag{10}$$

Where:

np	=	number of prototype aircraft
nta	=	number of test articles
W'AMPR	=	AMPR weight included in prototype
H_M	=	manufacturing manhours per pound of AMPR weight (determined from reference learning curve for manhours/lb. AMPR weight)
c_M	=	cost per manufacturing manhour
K_{ESCM}	=	escalation factor for manufacturing labor
W"AMPR	=	AMPR weight included in static and dynamic test articles

$$C_{MOH} = C_{MDL} \times O_M/100 \tag{11}$$

Where:

C_{MDL}	=	cost of manufacturing direct labor
O_M	=	manufacturing overhead factor

$$C_{MM} = (n_{eng} \times c_{eng} \times K_{eng}) + C_A + C_{OM_p} + C_{OM_{ta}} \tag{12}$$

Where:

n_{eng}	=	number of engines to be used in prototypes
c_{eng}	=	cost per engine
K_{eng}	=	fraction of new engine cost charged to the development program (depends on contract agreement with engine manufacturer)
C_A	=	cost of avionics in flight test prototype(s)
C_{OM_p}	=	cost of other manufacturing materials for prototype aircraft

C_{OMta} = cost of other materials for static test articles

$$C_{OMp} = c_{MM} \times K_{ESCMM} \times W'AMPR \qquad (13)$$

Where:

c_{MM} = aggregate cost per pound of manufacturing materials

K_{ESCMM} = cost escalation factor for manufacturing materials

$$C_{OMta} = c_{MM} \times K_{ESCMM} \times \Delta W''AMPR \qquad (14)$$

Quality Assurance Cost

Quality assurance costs may be formulated as follows:

$$C_{QA} = C_{QA_{DL}} + C_{QAOH} \qquad (15)$$

Where:

$C_{QA_{DL}}$ = cost of QA direct labor

C_{QAOH} = cost of QA overhead

$$C_{QA_{DL}} = fQA \times c_{QA} \times K_{ESC_{QA}} \left\{(W'AMPR \times H_M) + (W''AMPR \times H_M)\right\} \qquad 16)$$

Where:

fQA = proportion of QA manhours to manufacturing manhours in percent for a new aircraft development program

c_{QA} = cost per QA direct labor manhour

$$C_{QAOH} = C_{QADL} \times \frac{C_{QA}}{100} \qquad (17)$$

Where:

O_{QA} = QA overhead factor

Total Development and Certification Cost

The Total engineering, development, and certification cost is the sum of the above costs. This is formulates as follows:

$$C_{D/C} = C_E + C_T + C_M + C_{QA} \qquad (18)$$

Production Unit Cost

Production unit cost may either be determined as the cost of each article along the learning curve, with the appropriate escalations for labor and material costs, or it may be determined more approximately as an average cost over a given production quantity.

It is generally accepted that the improvement curve for light aircraft follows approximately an 85 percent slope until near 1000 production units. After 1000 units of production the slope gradually decreases to 90 to 95 percent due primarily to the incorporation of design improvements.

For the purposes of this analysis production unit cost has been estimated as an average over the first 3000 unit production quantity. Costs were calculated in terms of Mid-FY 77 dollars and held constant over the 3000 unit production quantity.

Total production unit average cost may be formulated as follows:

$$C_{PU} = C_{MM} + C_{MDL} + C_{MOH} + C_{QAOH} + C_{QADL} \quad (19)$$

Where:

C_{MM}	=	cost of manufacturing materials
C_{MDL}	=	cost of manufacturing direct labor
C_{MOH}	=	cost of manufacturing overhead
C_{QADL}	=	cost of quality assurance direct labor
C_{QAOH}	=	cost of quality assurance overhead

Each of the above costs is explained in the following formulations.

Manufacturing Materials Cost

Production manufacturing materials includes all airframe materials, aircraft systems, engines, avionics, interiors, exteriors, preparation costs, and production flight tests.

Manufacturing materials costs may be formulated as follows:

$$C_{MM} = (n_{eng} \times c_{eng}) + C_A + C_{OM} \quad (20)$$

Where:

n_{eng} = number of engines per aircraft

c_{eng} = average engine cost over the average engine set buy

C_A = cost of avionics

C_{OM} = cost of other manufacturing materials

$$C_{OM} = c_{MM} \times K_{ESCMM} \times W_{AMPR} \tag{21}$$

Where:

c_{MM} = aggregate cost per pound of manufacturing materials

K_{ESCMM} = cost escalation factor for manufacturing materials

W_{AMPR} = total AMPR weight

Manufacturing Direct Labor Cost

Production labor for this analysis is based on small aircraft industry experience projected to the cumulative average manhours on the order of 3000 units of production.

Manufacturing direct labor cost may be formulated as follows:

$$C_{M_{DL}} = (W_{AMPR} \times \frac{H_{Mca}}{W_{AMPRREF}} \times c_M \times K_{ESC_M}) \tag{22}$$

Where:

W_{AMPR} = total AMPR weight as previously described

H_{Mca} = cumulative average production manhours per pound of AMPR weight over given production quantity of reference aircraft

$W_{AMPRREF}$ = total AMPR weight of reference production aircraft

c_M = cost per production manhour

K_{ESC_M} = cost escalation factor for manufacturing

Manufacturing Overhead Cost

$$C_{MOH} = C_{MDL} \times O_M/100 \tag{23}$$

Where:

$C_{M_{DL}}$ = manufacturing direct labor cost from equation (22)

O_M = manufacturing overhead factor

Quality Assurance Direct Labor Cost

$$C_{QA_{DL}} = fQA \times \frac{C_{MDL}}{c_M K_{ESC_M}} \times c_{QA} \times K_{ESC_{QA}} \tag{24}$$

Where:

fQA = proportion of QA manhours to manufacturing manhours in percent for a production program

$C_{M_{DL}}$ = cost of manufacturing direct labor

c_M = cost per manufacturing direct labor manhour

c_{QA} = cost per quality assurance direct labor manhour

$K_{ESC_{QA}}$ = cost escalation factor for QA

Quality Assurance Overhead Cost

$$C_{QA_{OH}} = C_{QA_{DL}} \times O_{QA}/100 \tag{25}$$

Where:

$C_{QA_{OH}}$ = cost of quality assurance overhead

$C_{QA_{DL}}$ = quality assurance direct labor cost from equation (24)

O_{QA} = quality assurance overhead

Initial Pricing Estimate

The initial pricing estimate is obtained to facilitate comparison between alternate proposed products and between these products and the competition. These prices are expected at the production date of the average priced new product unit of the amortization base, or during any other year for which the cost analysis and pricing estimate relative dollar value is based.

The initial pricing estimate is the sum of the following values:

a. Production Unit Cost

b. Period Cost

c. Warranty Reserve

d. Gross Margin

e. Development Amortization

This may be formulated as follows:

$$P_I = C_{Pu} + C_p + C_{GM} + C_{D/A} \tag{26}$$

Where:

P_I	=	initial pricing estimate
C_{Pu}	=	average production unit cost over the amortization base
C_p	=	period cost per production unit
C_W	=	warranty reserve
C_{GM}	=	gross margin
$C_{D/A}$	=	development and amortization

Period cost, Cp, as used at GLC, includes all items essential to construction of the aircraft but not included in production unit cost. Period costs range from about 3 to 19 percent of production unit cost, and include marketing, field support, sustaining engineering, G&A, public relations, and corporate allocations. Sometimes it is desirable to load period costs more heavily against existing products that are selling well and lighten it for a new lower-priced product, or conversely, increase its burden on a new top-of-the-line product. Given these considerations period cost may be formulated as follows:

$$C_p = C_{Pu} \times f_{Cp} \times K_{ESCp} \times L \tag{27}$$

Where:

f_{Cp}	=	Period cost in percent
K_{ESCp}	=	escalation factor for period cost
L	=	loading factor where for a bottom-of-the-line product L may be as low as 0.25 and for a top-of-the-line product L may be as high as 1.5

For the last item, L, the distribution of the period cost loading factor over the total product line must balance out so that all period costs are covered in the overall product line pricing policy.

Warranty reserve cost is a rather arbitrary computation unless sufficient historical experience is available. Generally it can be related to cost and is valued at about 2 percent of factory cost for general aviation aircraft. State of the art improvements in product quality should hold this figure fairly constant, rather than requiring escalation as with direct expenses.

Gross margin, C_{GM}, as used herein, includes G & A, sales, commissions, distributors' allowances, sales margin, and corporate profit. On military or government programs G & A would be entered separately and the margin would be lower to suit the type of contract and the customer's acceptability.

Gross margin appropriate for most products would be 30% to 35% of list price. Generally, the larger the sales potential and the more competitive on price the particular market, the lower the gross margin. For a bottom-of-the-line airplane, where these factors are of paramount consideration, 20-25% might even be appropriate. Gross margin may be formulated as follows:

$$C_{GM} = (C_{Pu} + P_c + C_W) \frac{\left(\frac{GM}{100}\right)}{\left(1 - \frac{GM}{100}\right)} \tag{28}$$

Where:

GM = gross margin

The development amortization, $C_{D/A}$, is used to describe the cost of the development and certification program as a write-off against an amortization base. For internally funded programs, about three to four years of sales potential is considered normal for write-off of a development program. For outside funded programs the base would be the program first buy, or on a risk basis, the first X buys up to a three or four year production run. Management decisions would, of course, be involved in the latter case. Development amortization cost may be formulated as follows:

$$C_{D/A} = C_{D/C}/N_A \tag{29}$$

Where:

CD/C	=	development and certification cost
NA	=	number of aircraft in amortization base

Selling price is the sum of the above discussed costs, reserves, margins, and amortization write-off. The estimated selling price may be adjusted for other years by applying the appropriate escalation factor(s).

Estimated Cost of Operations

The method for estimating cost of operations is the same as that commonly used by the General Aviation Industry where an accounting is made for the following variable and fixed costs.

Variable Costs

- Fuel and oil
- Airframe and avionics maintenance reserves
- Mid-term hot section inspection (HSI) and parts reserve
- Engine overhaul reserves
- Parking/Landing fees and spare parts inventory

Fixed Costs

- Depreciation
- Crew compensation
- Insurance
  - Hull
  - Liability/Medical
- Storage and/or tie-down
- Navigation materials
- Airways tax

The methodology employed in estimating these costs is explained as follows:

<u>Variable Costs</u>

1. Fuel and Oil

   Gal./hr x $/gal

   Turbine fuel cost (Mid FY77) = $0.67/gal
   (w/o fuel additives)

   Oil cost per flight hour = $0.60

2. Airframe and Avionics Maintenance Reserves

   Maintenance at approximately 0.5 m-hrs/flt-hr.

Labor rate @ $14.00/hr.

Maintenance cost = 0.5 × 14.00 = $7.00/flt-hr.

3. Mid-Term HSI and Parts Reserve

Cost based on $3.50/lb-thrust

500 lb-t × 3.50 = $1750

At 200 hrs/yr., TBO = 2575 hrs.

HSI at 1250 hrs.

$$\text{cost/hr} = \frac{1750}{2575} = \$1.36/\text{hr.}$$

4. Engine Overhaul Reserves

Cost of overhaul = 40% of original cost

TBO = f(annual utilization)

5. Parking/Landing Fees and Spare Parts Inventory

Adjusted to Mid-FY77 costs

Fixed Costs

1. Depreciation

Assumed to be straight line over 8 years decreasing to 20 percent original purchase price.

2. Crew

Not applicable

3. Insurance

Hull: Equal to 0.8% × original purchase price

Liability/Medical: Adjusted to Mid-FY77 costs

4. Storage

Adjusted to Mid-FY77 costs

5. Navigation Materials

Estimated price

6. Airways Tax

Prevailing amount

APPENDIX D - CRITIQUE OF GASP COST METHODOLOGY

The cost analysis methodology in the GASP program for estimating flyaway factory (FAF) costs was derived from correlated statistical data obtained by survey of many different manufacturers and their products. The resulting cost estimation relationships (CER's) included:

Inputs

Weight/Speed

Power/Propulsion Type

Block Fuel and Time

Cost coefficients

Solutions

Flyaway Cost	Operating Cost
Labor	Variable
Materials	Fixed
Purchased Equipment	Utilization
Mark-ups	

Output

Flyaway Cost/Breakdown

Operating Cost vs. Utilization

Following collection of cost/prices and physical characteristic data correlation was determined by NASA through regression analysis and other means. With this information, cost estimating relationships were determined for incorporation as model subroutines in the general design computer programs.

Examination of each of the cost estimating relationships applicable to a small turbofan powered airplane was conducted with comparisons made to methods employed by Gates Learjet, and to current (FY77) cost estimating coefficients. The results of this examination are discussed for each of the GASP CER's in the following.

Manufacturing Labor Manhours and Cost

Two relationships were developed for GASP - one for light aircraft and one for heavier high performance and turbofan powered aircraft. The one for turbofan powered aircraft is:

$$DMLH = WSP \times (3.9 \times 10^{-10} \times WSP + 2.5 \times 10^{-3})$$

$$CSML = DMLH \times ALR \times CLF$$

Where:

DMLH	= Manufacturing direct labor manhours
WSP	= WE × VCR_{MPH}
WE	= Aircraft empty weight
CSML	= Cost of manufacturing direct labor
ALR	= Average manufacturing labor rate
CLF	= Complexity factor

Solution of this expression for an iarcraft of empty weight equal to 1026.8 lbs. (WE of contemporary design aircraft) yields a manufacturing performance of 0.42 m-hrs/lb. The best performance of light aircraft manufacturers in large run production is estimated to be not less than a cumulative average of 0.60 m-hrs/lb. over 2000-3000 units and about 0.70 m-hrs/lb at about 1500 units. Model changes and product improvements cause a flattening of the improvement curve over about 1000 production units on small aircraft with the curve changing from approximately 85 percent to 90 to 95 percent as improvements are incorporated.

The major concerns with this relationship is that it is based only on relatively large quantity production and appears to underestimate manufacturing manhours by 30 to 40 percent.

Manufacturing Overhead Percent and Cost

The relationships in GASP for estimating manufacturing overhead percent and cost are as follows:

$$OHML = (7 \times 10^{-8} \times WSP) + 1.31$$

$$CSOH = OHML \times CSML$$

Where:

OHML = Manufacturing overhead in percent of direct labor hours

WSP = WE x maximum cruise speed in mph

CSOH = Cost of manufacturing overhead in dollars

CSML = Cost of manufacturing direct labors in dollars

Solution of OHML for both small and medium sized high performance aircraft yields an overhead percentage that is, by current practices, too low. Recent government guidelines have required full absorption of some sustaining costs like tooling and others to be included in manufacturing overhead which has raised the level to the range of 155 to 165 percent of manufacturing direct labor costs. Thus, a reappraisal is needed to change this CER to reflect current accounting practices. In this regard manufacturing overhead should include an accounting for:

Manufacturing Management and Supervision

Training

Direct Manufacturing Services

Production Control

Manufacturing Planning

Industrial Engineering

Manufacturing Engineering

Quality Assurance

Sustaining Tooling

Facilities Administration

Materials Administration

Of these overhead costs, the inclusion of sustaining tooling is responsible for 90-95% of the increase from 130-135 percent to the 155-165 percent range. Changes would also be necessary in the CER to estimate sustaining costs which will be discussed later.

Manufacturing Material Cost

The CER for estimating light aircraft manufacturing materials cost is:

$$CSMM = \left\{(1.5 \times 10^{-4}) \times WE + .38\right\} \times WE$$

Where:

CSMM = Cost of manufacturing materials

Solution of this equation for a WE of 1026.8 lbs. yields \$0.534/lb. of WE. Inflation between 1970 and FY-77 would increase this cost factor by approximately 61 percent to \$0.86/lb. of WE. However, actual cost of raw manufacturing materials in FY77 were about \$4-\$5 per pound for light aircraft.

A reappraisal of manufacturing material cost is apparently needed along with CER changes to permit cost rate fluctuations with inflation.

Airframe Fabrication Cost

The CER for summing manufacturing cost of airframes is simply:

CSAFF = CSML + CSOH + CSMM

Where:

CSAFF = Airframe Fabrication Cost

Original Equipment Factor for Engines and Propeller - List Price Cost

This CER is not applicable to turbofan powered aircraft.

Engine Cost

The CER for turbofan engine cost is a product as follows:

CSENG = \$Lb.T × BHP1

BHP1 is this equation is taken as engine maximum seal level static thrust.

Total Propulsion Cost

The CER for propulsion cost is given as follows:

CSPPUL = (CSENG × YNE) + (CSPP × XP)

Where:

CSPPUL = Total propulsion cost

CSENG = Engine cost

YNE = Number of engines/aircraft

CSPP = Propeller cost

XP = Number of propellers/aircraft

The latter two items, of course, do not apply to turbofan powered aircraft.

Other Equipment Cost

The CER for other equipment includes the cost of purchased equipment, excluding propulsion equipment. The CER is given as follows:

$$CSOEQ = 9.6 \times 10^{-7} \times (WSP)^{1.698}$$

Where:

CSOEQ = Cost of other equipment

WSP = Empty weight times maximum cruise speed in MPH

For the selected contemporary design aircraft this CER yields a CSOEQ = \$704. Inflation over the period from 1970 to Mid FY77 amounts to at least 60% which would increase CSOEQ to \$1126. However, this cost is still underestimated by 25 percent or more relative to FY-77 typical costs and therefore it appears that this CER needs a reappraisal, as well as provisions to permit changes for cost inflation.

Total Equipment Cost

The CER for total equipment cost is the sum of propulsion and other equipment costs as follows:

$$CSTEQ = CSPPUL + CSOEQ$$

Direct Manufacturing and Equipment Cost

This CER is the total of airframe fabrication and equipment cost as follows:

$$CSDME = CSAFF + CSTEQ$$

Engineering, Tooling, Sales, and Administrative Factor

This CER is expressed as follows:

$$ETSGA = 0.1669 \ (WE)^{0.08743}$$

Where:

ETSGA = Fraction of CSDME to be added for sustaining engineering and tooling, sales and G&A cost

For a WE = 1.026.8 lbs., ETSGA is calculated to be 0.306. Changes in accounting practices and guidelines has required that sustaining tooling be covered in manufacturing overhead cost; therefore, this factor should be reduced accordingly. These costs are approximately the same as that described as Period Costs in the contemporary method described in Appendix C.

Total Factory Cost

The CER for total factory cost is the following sum:

CSMANF = CSDME + (ETSGA x CSDME)

Factory Profit Goal and Dealer's Cost

The CER for estimating factory profit goal is:

$PROFG = (2.33 \times 10^{-5} \times WE) + .066$

This yields a profit goal of approximately 9 percent for a WE = 1026.8 lbs. which appears reasonable for this size aircraft. For larger, higher performance aircraft, however, this CER will yield factory profit goals of 25 percent or more which does not appear reasonable in highly competitive markets of medium size business jet aircraft. Therefore, a possible alternative CER should be considered with user selected input factors for profit goals.

Distributor and Dealer Mark-up Trend

The CER for estimating mark-up is

$DDMARK = 0.1695 \times (WE)^{0.08743}$

This yields a markup of 31 percent for a WE = 1026.8 lbs. which also appears reasonable for small size aircraft. This CER estimates dealer markup as a percentage of factory price which is not common practice. In actual practice the markup is based on percent of dealer price. As in the CER for estimating factory profit, consideration should be given to provision of a user selected input factor for a markup goal.

Total Flyaway Factory List Price

This CER is expressed as follows:

$$CSAFAF = CSDLR + (DDMARK \times CSDLR)$$

The value of CSAFAF is generally expressed as a unit cost without add-on equipment. Any additional equipment will also have markup values over the factory or dealer cost of provisioning on the airplane.

Operating Costs

Variable Costs

A. Fuel and Oil

Cost of fuel and oil per hour of operation is based on estimated block fuel/oil consumption for an average mission range. The CER is given as follows:

$$CSFL = GPH \times CSFG$$

Where:

GPH	=	Fuel consumption rate in gal./hr.
CSFG	=	Fuel cost per gallon
CSOL	=	\$0.20 average cost/hr.
CSHPOL	=	CSFL + CSOL

Actual oil cost has escalated to about 60 cents per hour in FY-77 for equivalent consumption. Cost of fuel has also escalated and in Mid FY-77 was about 67 cents per gallon including taxes, but not including additives. Additives would add 2-3 cents per gallon.

B. Inspection and Maintenance

Cost of inspection and maintenance is expressed as follows:

$$AIC = \frac{CINP}{HRI}$$

Where:

AIC	=	Cost of inspection and maintenance per flight hour
CINP	=	Cost of inspection in dollars (If not input default = \$1500)
HRI	=	Hours between inspection (If not input default = 100 hrs)

C. Reserve for Engine Overhaul

Cost of engine overhaul is expressed as follows:

$$OHC = \frac{ENP \times TSLS \times OHR}{TBO} \quad \text{for turbofan (NTYE = 7)}$$

Where:

OHC	=	Cost of overhaul per flight hour
ENP	=	Number of engines
TSLS	=	Seal level static thrust
OHR	=	Cost of overhaul per pound of thrust (If not input default = \$5.5/lb.thrust)
TBO	=	Time between overhaul (Input)

D. Parking/Landing Fees, Spare Parts Inventory

These costs must be accounted for by inputting a value for CMV (increment to hourly operating costs) into the GACOSTroutine. In early CY77 these costs amount to approximately \$0.85 for parking/landing fees and \$0.32 for spare parts inventory per hour of operation for Category I airplanes.

Fixed Costs

A. Depreciation

$$CSYDP = (CSFAF + CSOPT)\frac{80}{100} \times \frac{1}{8}$$

Where:

CSYDP	=	Depreciation cost per annum
CSFAF	=	Dealer price at factory
CSOPT	=	Price of added factors

Depreciation is assumed to diminish the value down to 20 percent over a period of eight years. This CER is a modified version of the original where the airplane original value was depreciated 100 percent in 20 years. The CER as expressed is definitely more appropriate.

Depreciation cost per flight hour is a function of annual utilization.

CSHDP = CSYDP/AU

Where:

AU = Annual utilization in hours

B. Insurance

Hull Insurance

CAT I - HINS = HIR x CSFAF

Where:

HIR = Hull Insurance rate in percent (default value = 2%)

Liability Insurance

CAT I - LINS = CLI

Where:

CLI = Cost of liability insurance (default value = $215)

Liability insurance has increased in cost due to inflation and in early CY 77 would cost about $325 annually.

C. Storage

Storage costs cover the cost of tie-down and hanger and are accounted for by the input variable SRPM in GACOST. Default value of SRPM is zero. In early CY77 storage costs for CAT I airplanes would be approximately $450 per year.

D. Pilot (Crew) Cost

This cost is not applicable to small aircraft.

E. Miscellaneous Fixed Costs

This cost can include annual expenditures for maps, manuals, and other incidental items. It is an input CMF in dollars per year whose default value is zero.

For CAT I airplanes this cost is approximately \$100/year in early CY77.

F. FAA Use Tax

For gross weights $\leq$ 2500 lbs., this cost is as follows:

CSTAX = \$25/year.